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REPORT No. 126

STRUCTURAL PROOF TEST PROGRAM

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Report Number 126

STRUCTURAL PROOF TEST PROGRAM

U.S. Army XV-5A Lift Fan
Research Aircraft

Advanced Engine and Technology Department
Flight Propulsion Division
General Electric Company
Cincinnati, Ohio 45215

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INTRODUCTION

The structural proof test program to be accomplished for the U.S. Army XV-5A Lift Fan Research Aircraft is presented in this report. The structural static test program is a part of the basic ground tests as required under Contract No. DA-44-177-TC-715. A general outline of the program for all basic ground tests is given in Basic Ground and Full Scale Wind Tunnel Test Program.

The test program described herein is designed to demonstrate integrity of aircraft structure and the information presented will be used to establish detail test procedures.

The particular flight and landing load conditions to be simulated during test have been derived from structural analysis using the conditions specified in the Airplane Structural Design Criteria and have been found critical. A detailed listing of all test data requirements is given herein. The particular loads and reactions to be applied to the airframe and the major components are also given. Along with the strength tests, some additional control system tests are to be performed and are also described.

II. TEST ARTICLE

The proof test article shall be the complete Number One airplane (Army Serial No. 62-4505) which is a flight article, minus certain items not essential for the tests, for example:

> Electrical equipment, hydraulic lines, seats, gas generators, engine compartment doors, nose fan, nose fan ducting, nose fan louvers, fuselage access fairing, fuel tanks, cross-over ducts, tail pipes, control actuators, pitch fan doors, thrust reverser doors, parachute compartment fairing, pitot tube and/or sensing devices, fuselage center-section side panels, and engine inlet.

Some items may be on the airplane which are not required for the tests, if omission would interfere with the normal production sequence required to complete the airplane.

Other items to be omitted from the airplane are those which are to be tested separately in jig fixtures. These items are:

> Canopy (to be tested to ultimate), flaps, wing fan, and fan door support structure. The rudder is also to be tested in a separate fixture but it is to be installed on the airplane during empennage tests to check for control surface binding.

Both nose and main landing gears are a part of the test article and shall be static tested on the airplane.

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III. GENERAL TEST PROCEDURE

Each of the tests shall be conducted in accordance with the detailed description given in this report. All applied test loads shall be imposed to simulate, nearly as possible, the actual loads encountered at the critical conditions. Tests shall not necessarily be conducted in the order given in this report.

Test equipment shall include hydraulic cylinders, tension pads and whiffle-trees, strain gages, recorders, deflection scales, shot bags, and a hydraulic pressure regulating unit.

The hydraulic cylinders shall be connected to a single regulating unit for even distribution of load. The tension pad whiffle-tree arrangement shall be used for applying the proper loads to the wing, with the line of action of the test loads acting at the specified angles with the wing plane to simulate the forward or aft component of load occuring in the condition. The stresses at critical points shall be measured by strain gages, and the deflections measured by hanging scales or equivalent. Only those strain gages specified in a certain area shall be read and recorded for that particular test. Primary purpose of the strain gages is to monitor strain, as limit load is approached, and to detect any unpredicted internal load distribution resulting in excessive strain.

Unless otherwise specified, the load increments in percent of design limit load shall be as follows:

0%	-	-	-	-	-
20%:	Record	strain	and	deflection	readings.
40%:	"	"	"	"	"
20%:	"	"	"	"	"
60%:	"	"	"	"	"
20%:	"	"	"	"	"
80%:	"	"	"	"	"
20%:	"	"	"	"	"
90%:	"	"	"	"	"
20%:	"	"	"	"	"
100%:	"	"	"	"	"
20%:	"	"	"	"	"

No yielding of structure should occur at 100% limit load. If during the test, yielding does appear imminent, or some other unforeseen difficulty arises, the test shall be stopped at the discretion of the responsible structures engineer and any required change(s) made. The test shall be resumed by starting from zero load.

For all tests, unless otherwise specified, the 100% limit load shall be held for a time not to exceed three minutes, during which time the structure shall be inspected for adverse characteristics, and required strain and deflection values

shall be recorded. Inspection of structure while it is under load must be limited to those areas that can be seen without danger to test personnel.

IV. STRAIN GAGE LOCATIONS

TABLE 1

STRESS CODE	INSTRU. CODE	GAGE LOCATION	MEAS.	NO. OF GAGES & TYPE	FLT. TEST	STATIC TEST
		HORIZONTAL STABILIZER:				
S-101	S-43	Axial Actuator Load	Axial	1-144	X	X
S-102	S-36	L.H. Fwd. Upper Spar Cap, B.L.6	"	1-141		X
S-103		L.H. Fwd. Lower Spar Cap, B.L.6	"	"		X
S-104	S-45	L.H. Center Spar, Upper Cap, B.L.6	"	"		X
S-105		L.H. Center Spar, Lower Cap, B.L.6	"	"		X
S-106	S-40	L.H. Aft Spar, Upper Cap, B.L.6	"	"		X
S-107		L.H. Aft Spar, Lower Cap, B.L.6	"	"		X
S-108	S-38	L.H. Rib, Upper Cap, Sta. 501, B.L.4	"	"		X
S-109		L.H. Rib. Lower Cap, Sta. 501, B.L.4	"	"		X
S-110	S-99	L.H. Center Spar, Upper Cap, B.L.30	"	"		X
S-111		L.H. Center Spar, Lower Cap, B.L.30	"	"		X
S-112	S-100	L.H. Rear Spar, Upper Cap, B.L.30	"	"		X
S-113		L.H. Rear Spar, Lower Cap, B.L.30	"	"		X
		VERTICAL FIN:				
S-201	S-34	Vertical Fin Spar (L.H.Cap at W.L. 200)	Axial	1-144	X	X
S-202	S-37	Vertical Fin Spar (R.H. Cap at W.L. 200)	"	"	X	X
S-203	S-50	Fwd. Spar, L.H. Cap, V.S. Sta 13.4	"	1-141		X
S-204		Fwd. Spar, R.H. Cap, V.S. Sta 13.4	"	"		X
S-205	S-53	Center Spar, L.H. Cap, V.S. Sta.13.4	"	"		X
S-206		Center Spar, R.H. Cap, V.S. Sta.13.4	"	"		X
S-207	S-56	Aft Spar, L.H. Cap, V.S. Sta.13.4	"	"		X
S-208		Aft Spar, R.H. Cap, V.S. Sta. 13.4	"	"		X

TABLE 2

STRAIN GAGE LOCATIONS

STRESS CODE	INSTRU. CODE	GAGE LOCATION	MEAS.	NO. OF GAGES & TYPE	FLT. TEST	STATIC TEST
		FORWARD FUSELAGE				
S-301	S-57	Upper Longeron, F.S.91, L.H.	Axial	1-141		X
S-302	S-58	Upper Longeron, F.S.165, L.H.	"	"		X
S-303	S-59	Upper Longeron, F.S.214, L.H.	"	"		X
S-304	S-65	Lower Longeron, F.S.91, L.H.	"	"		X
S-305	S-66	Lower Longeron, F.S.165, L.H.	"	"		X
S-306	S-67	Lower Longeron, F.S.214, L.H.	"	"		X
		AFT FUSELAGE				
S-401	S-60	Upper Longeron F.S.287, L.H.	Axial	1-141		X
S-402	S-61	Upper Longeron F.S.287, R.H.	"	"		X
S-403	S-62	Upper Longeron F.S.316, R.H.	"	"		X
S-404	S-68	Lower Longeron F.S.300, L.H.	"	1-144	X	X
S-405	S-69	Lower Longeron F.S.300, R.H.	"	1-141		X
S-406	S-70	Lower Longeron F.S.316, R.H.	"	"		X
S-407	S-71	Lower Longeron F.S.400, L.H.	"	"		X
S-408	S-72	Lower Longeron F.S.400, R.H.	"	"		X
S-409	S-73	Top Longeron F.S.315	"	"		X
S-410	S-74	Top Longeron F.S.380	"	"		X
S-411	S-75	Frame Skin Flg., F.S.389.7, W.L.90, B.L. 22.5	"	"		X
S-412	S-76	Frame Skin Flg., F.S.389.7, W.L.113, B.L. 23.2	"	"		X
S-413	S-101	Frame Skin Flg., F.S.377.25, W.L.119, B.L. 22.5	"	"		X
S-414	S-102	Frame Skin Flg.,F.S.377.25, W.L.150, B.L. 0	"	"		X
S-415	S-79	Side Skin Shear, W.L.100, F.S.287, L.H.	Shear	1-121-R3A		X
S-416	S-113	Frame Flanges F.S.287, W.L.96, B.L. 11.3	Axial	1-141		X
S-417	S-114	Frame Flanges F.S.287, W.L.105, B.L. 11.3	"	"		X
S-418	S-115	Frame Flanges F.S.287, W.L.131.4, B.L. 15.5	"	"		X
S-419	S-116	Frame Flanges F.S.287, W.L.133.5, B.L. 15.5	"	"		X

TABLE 3

STRAIN GAGE LOCATIONS

STRESS CODE	INSTRU. CODE	GAGE LOCATION	MEAS.	NO. OF GAGES & TYPE	FLT. TEST	STATIC TEST
		<u>CENTER FUSELAGE</u>				
		Ref. Dwg. 143F009, Sheet 1				
S-501	S-80	Space Frame Member 8-28	Axial	1-141		X
S-502	S-81	Space Frame Member 9-28	"	"		X
S-503	S-82	Space Frame Member 25-30	"	"		X
S-504	S-83	Space Frame Member 26-29	"	"		X
S-505	S-85	Space Frame Member 9-31	"	"		X
S-506	S-86	Space Frame Member 25-31	"	"		X
S-507	S-87	Space Frame Member 26-31	"	"		X
S-508	S-88	Space Frame Member 11-31	"	"		X
S-509	S-89	Space Frame Member 8-13	"	"		X
S-510	S-91	Space Frame Member 11-14	"	"		X
S-511	S-92	Space Frame Member 11-26	"	"		X
S-512	S-93	Space Frame Member 8-25	"	"		X
S-513	S-94	Bulkhead Frame, AL.,F.S.214.00(1-25)	"	"		X
S-514	S-95	Front Wing Spar, Lwr Cap.(25-26)	"	"		X
S-515	S-96	Bulkhead Frame, AL.F.S.214.00(2-26)	"	"		X
S-516	S-97	Bulkhead Frame, AL.F.S.214.00(1-2)	"	"		X
S-517	S-103	Space Frame Member 14-31	"	"		X
S-518	S-104	Space Frame Member 4-25	"	"		X
S-519	S-105	Space Frame Member 5-26	"	"		X
S-520	S-106	Space Frame Member 3-25	"	"		X
S-521	S-107	Space Frame Member 6-26	"	"		X
S-522	S-108	Space Frame Member 9-13	"	"		X
S-523	S-109	Space Frame Member 10-14	"	"		X
S-524		Space Frame Member 4-39	"	"		X
S-525		Space Frame Member 5-40	"	"		X
S-526		Space Frame Member 9-43	"	"		X
S-527		Space Frame Member 10-44	"	"		X
S-528		Space Frame Member 2-4	"	"		X
S-529		Space Frame Member 9-14	"	"		X
S-530		Space Frame Member 17-35	"	"		X
S-531		Space Frame Member 17-36	"	"		X
S-532	S-110	Front Wing Spar, Upper Cap.F.S.214.00	"	"		X
S-533	S-112	Front Wing Spar, End Stiff.B.L.24.0	"	"		X

TABLE 4

STRAIN GAGE LOCATIONS

STRESS CODE	INSTRU. CODE	GAGE LOCATION	F.S.	B.L.	W.L.*	MEAS.	NO. OF GAGES & TYPE	FLT. TEST	STATIC TEST
		WING							
		Fwd. Spar Ftg.							
S-601	S-1	L.H. Upper Cap	214.3	25.50	5.10	Axial	1-141		X
S-602		L.H. Upper Cap	215.0	25.50	6.00	"	"		X
S-603	S-2	L.H. Lower Cap	215.3	26.175	-2.00	"	"		X
S-604		L.H. Lower Cap	215.3	26.175	-3.50	"	"		X
		Fwd. Spar Caps							
S-605		L.H. Upper	214.0	40.0	5.6	"	1-144	X	X
S-606		L.H. Lwr.	214.0	40.0	-3.6	"	"	X	X
S-607	S-3	L.H. Upper	214.9	61.0	6.0	"	1-141		
S-608	S-4	L.H. Lwr.	214.9	61.0	-4.0	"	"		X
S-609	S-5	L.H. Upper	226.0	88.5	6.7	"	"		X
S-610	S-6	L.H. Lwr.	226.0	88.5	-3.5	"	"		X
		Fwd. Spar Splice							
S-611	S-7	L.H. Upper	232.9	112.1	6.3	"	"		X
S-612	S-8	L.H. Lower	233.3	112.9	-2.7	"	"		X
		Fwd. Spar Caps							
S-613	S-26	R.H. Upper	214.0	40.0	5.6	"	1-144	X	X
S-614	S-27	R.H. Lower	214.0	40.0	-3.6	"	"	X	X
		Fwd. Spar Web							
S-615	S-18	L.H. ℄		28.8	1.7	Shear	1-121-R3A		X
S-616		L.H. Lwr. Lug	Rear	28.8	0	"	"		X
S-617		L.H. ℄	Face	39.8	1.0	"	1-124-R3A	X	X
S-618	S-19	L.H. ℄	of	59.0	1.10	"	1-121-R3A		X
S-619		L.H. ℄	Web	88.5	1.35	"	"		X
S-620		L.H. ℄		112.3	1.8	"	"		X
S-621	S-30	R.H. ℄		39.8	1.0	"	1-124-R3A	X	X
		L.H. Main Rib							
S-622	S-9	Front Spar Attach., Upr.	235.0	100.75	8.2	Axial	1-141		X
S-623	S-10	Front Spar Attach., Lwr.	23[illegible].0	100.75	-4.9	"	"		X
S-624		F.S. Attach, L.E., Upr.	223.7	100.75	5.8	"	"		X
S-625		F.S. Attach, L.E., Lwr.	223.7	100.75	-2.6	"	"		X
S-626	S-98	Web at F.S.	228.7	100.75	1.75	Shear	1-121-R3A		X
S-627		Web at R.S.	295.3	100.75	1.10	"	"		X

*Relative to W.L.100

TABLE 5

STRAIN GAGE LOCATIONS

STRESS CODE	INSTRU CODE	GAGE LOCATION	F.S.	B.L.	W.L.*	MEAS.	NO. OF GAGES & TYPE	FLT. TEST	STATIC TEST
		WING (cont'd)							
		<u>Rear Spar, L.H.</u>							
S-628		Ftg., Upr. Cap	296.8	25.5	5.0	Axial	1-141		X
S-629	S-11	Ftg., Upr. Cap	297.5	25.5	5.7	"	"		X
S-630		Ftg., Lwr. Cap	297.8	26.175	-2.0	"	"		X
S-631	S-12	Ftg., Lwr. Cap	297.8	26.175	-3.5	"	"		X
S-632		Upr. Cap	297.2	29.0	6.2	"	"		X
S-633		Lwr. Cap	296.8	29.0	-2.5	"	"		X
S-634		Lwr. Cap	297.5	29.0	-4.0	"	"		X
S-635		Upr. Cap	297.0	39.6	5.6	"	1-144	X	X
S-636		Lwr. Cap	297.0	39.6	-3.4	"	"	X	X
S-637	S-13	Upr. Cap	296.5	61.0	6.2	"	1-141		X
S-638	S-14	Lwr. Cap	296.5	61.0	-4.0	"	"		X
S-639	S-15	Upr. Cap	296.5	99.0	6.2	"	"		X
S-640	S-16	Lwr. Cap	296.5	100.2	-4.0	"	"		X
S-641		Splice, Upr. Cap	297.2	111.8	6.0	"	"		X
S-642		Splice, Lwr. Cap	297.2	112.5	-2.3	"	"		X
		<u>Rear Spar, R.H.</u>							
S-643	S-28	Ftg. R.H. Upr.	296.5	39.6	5.6	"	1-144	X	X
S-644	S-29	Ftg. R.H. Lwer	296.5	39.6	-3.4	"	"	X	X
		<u>Rear Spar Webs, L.H.</u>							
S-645		Top		29.0	3.2	Shear	1-121-R3A		X
S-646	S-20	℄	Rear	29.0	1.5	"	"		X
S-647		W.L.0	Face	"	0	"	"		X
S-648		Bottom	of	30.4	-1.0	"	"		X
S-649		℄	Web	39.6	1.1	"	1-124-R3A	X	X
S-650		"		57.5	1.1	"	1-121-R3A		X
S-651		"		100.25	1.1	"	"		X
S-652		"		112.3	1.8	"	"		X
S-653	S-31			39.0	1.1	"	1-124-R3A	X	X

*Relative to W.L. 100

TABLE 6

STRAIN GAGE LOCATIONS

STRESS CODE	INSTRU CODE	GAGE LOCATION	F.S.	B.L.	W.L.	MEAS.	NO. OF GAGES & TYPE	FLT. TEST	STATIC TEST
		WING (cont'd)							
		Leading Edge, L.H.							
S-654	S-17	Stringer	190.0	28.1	.5	Axial	1-141		X
S-655		Rib Cap, Upr.	211.3	57.0	5.4	"	"		X
S-656		Rib Cap, Lwr.	211.3	57.0	-3.5	"	"		X
S-657	S-23	Upper Skin	209.0	25.5	5.50	Shear	1-124-	X	X
							R3A		
S-658	S-24	Lwr. Skin	209.0	25.5	-3.8	"	"	X	X
S-659		Upper Skin	202.9	63.8	1.4	"	1-121-		X
							R3A		
S-660		Lwr. Skin	204.5	63.8	-1.0	"	"		X
		Leading Edge, R.H.							
S-661		Upper Skin	209.0	25.5	5.5	Shear	1-124-	X	X
							R3A		
S-662		Lwr. Skin	209.0	25.5	-3.8	Shear	1-124-	X	X
							R3A		
		143P035-53							
		Fan Supt. Links					(600°F)		
S-663		L.H. Link, Upper End				Axial		X	X
S-664		R.H. Link, Upper End				"		X	X

V. PROOF TESTS

TEST NO. 1

Condition

4-g symmetrical maneuver F-1, positive low angle of attack, V = 500 K. Sea level, zero pitching acceleration.

Critical Structure

This is a critical condition for the wing basic structure and its attachment to the fuselage.

Test Procedure

The airplane shall be supported as shown in Figure 1. Wing fans are not to be installed. Reactions need not be measured.

Increments of limit load as specified in the general test procedure shall be applied through pads to the L.H. and R.H. wings at the load points shown. The test loads were derived from the wing panel point loads used in the stress analysis and provide the proper shear moment, and torsion. The front and rear fan fittings are to be installed and used as loading points to simulate aerodyanmic and inertia forces from the fan-door assembly. Resultant test loads all act aft at an angle of 10.5 degrees with the vertical to simulate both normal and drag components occuring in the condition. The ailerons are to be installed and checked while wing is under 100% limit load to insure that they do not bind.

Measurements

Measurements shall be made of the position of the aileron valve driving linkage at each incremental loading condition to obtain valve travel induced by deflection of structure. The lateral control system shall be rigged in the aileron undrooped condition, and the cockpit control stick restrained in a neutral position. Equivalent valve travel shall be derived in this manner for both the right and left aileron systems.

Readings of all wing strain gages shall be recorded.

The following deflections shall be recorded:

(1) Point 100, L.H. Wing, Vertical deflection.
(2) Point 100, R.H. Wing, Vertical deflection.
(3) Point 102, L.H. Wing, Vertical deflection.
(4) Point 102, L.H. Wing, Aft deflection.
(5) Point 102, R.H. Wing, Vertical deflection.
(6) Point 102, R.H. Wing, Aft deflection.

(7) Point 106, L.H. Wing, Vertical deflection.
(8) Point 108, L.H. Wing, Vertical deflection.
(9) Point 113, L.H. Wing, Vertical deflection.
(10) Point 114, L.H. Wing, Vertical deflection.
(11) Point 115, L.H. Wing, Vertical deflection.
(12) Point 119a, L.H. Wing, Vertical deflection.
(13) Point 122a, L.H. Wing, Vertical deflection,
(14) Front Spar Attach Point, L.H. Wing, Vertical deflection.
(15) Front Spar Attach Point, L.H. Wing, Aft deflection.
(16) Front Spar Attach Point, R.H. Wing, Vertical deflection.
(17) Front Spar Attach Point, R.H. Wing, Aft deflection.
(18) Rear Spar Attach Point, L.H. Wing, Vertical deflection.
(19) Rear Spar Attach Point, R.H. Wing, Vertical deflection.

TABLE 7

TEST NO. 1 WING RESULTANT LOADS, L.H. & R.H.

Load Point Number	Location B.L.	Location F.S.	Limit Resultant Load (Resultant acts up and aft at 10.5° with vertical.)	
100	168.0	277.0	209#	
101	169.0	287.1	0	
101a	160.4	284.2	223	
102	169.0	294.0	556	
102a	160.4	294.0	0	
103	151.7	264.7	508	
104	151.7	280.6	215	
104a	143.0	277.5	299	
105	151.7	294.0	277	
105a	143.0	294.0	315	
106	134.3	251.7	689	
107	134.3	274.1	363	
107a	125.6	270.9	395	
108	134.3	294.0	322	
108a	125.6	294.0	329	
109	116.9	238.7	763	
110	116.9	267.6	427	
110a	108.2	264.3	413	
111	116.9	294.0	336	
11a	108.2	294.0	343	
112	-	-	0	
113	100.75	226.5	1040	Hoist Fitting
114	100.8	261.5	399	
115	100.8	294.0	310	
116	80.7	220.2	1304	
117	-	-	0	
118	61.0	204.8	485	
119a	61.0	217.4	972	Fwd. Wing Fan Ftg.
122a	61.0	294.5	1450	Aft Wing Fan Ftg.
123	42.5	214.0	771	
124	42.5	296.5	-109	Down
125	-	-	0	
126	-	-	0	
Total	104.8	257.4	13604#	

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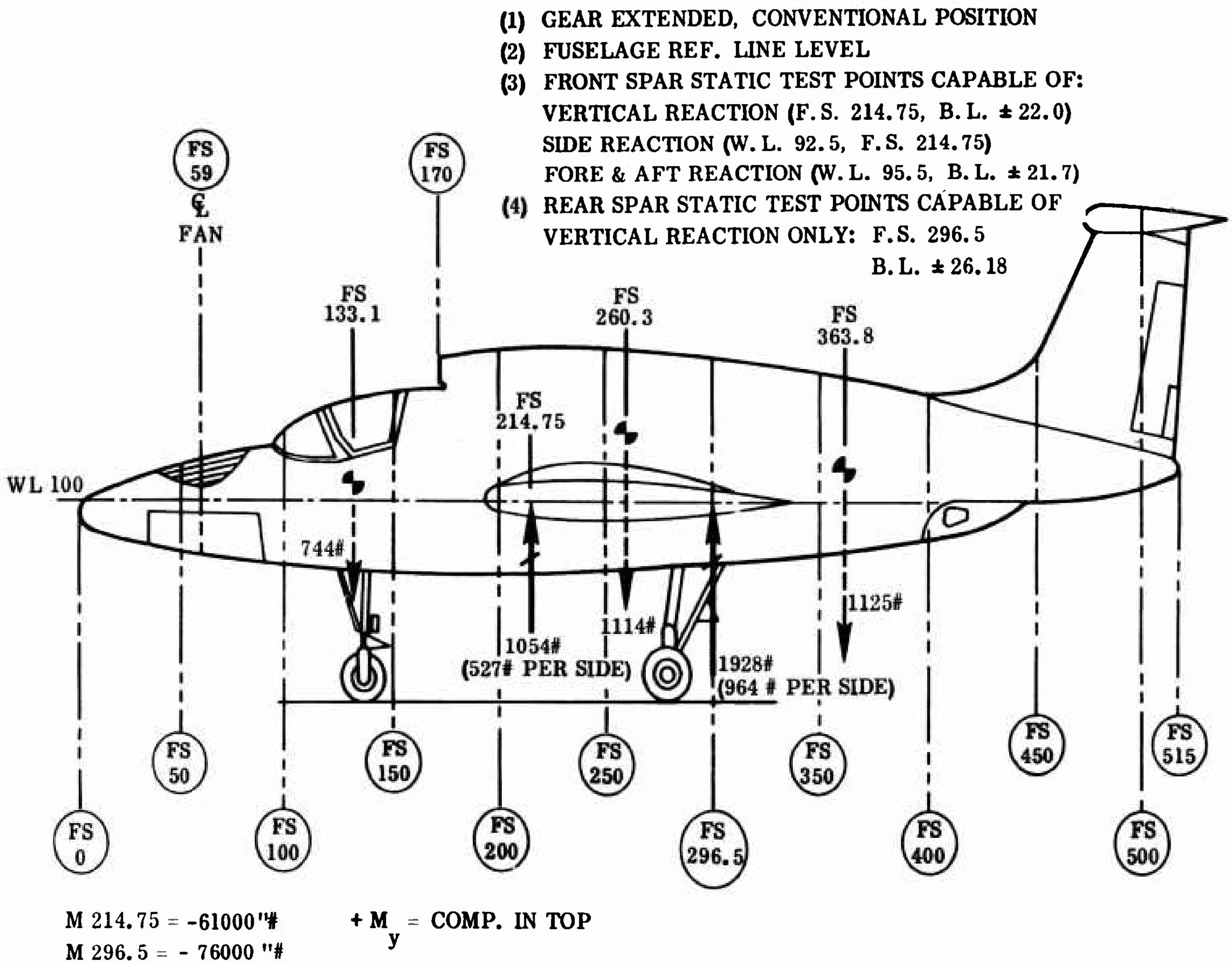


Figure 1 Estimated Test Article Weight Distribution for Support at Front Spar and Rear Spar Static Test Points

TEST NO. 2

Condition

2-Wheel,Tail Down Landing, Dynamic Spring Back

Critical Structure

This is one of the critical conditions for the main landing gear; local support structure and fittings, and the fuselage forward of F.S. 316, including the center section or space frame.

Test Procedure

The airplane shall be mounted as shown in Figure 2. The reactions at the jacking fittings shall be measured, ($\pm$10% accuracy).

Increments of limit load as specified in the general test procedure shall be applied. Fuselage loads which do not come at hard points or fittings shall be applied through straps riveted to the fuselage at frame locations shown. Main gear vertical and forward loads shall be applied at the axles with oleos fixed in the 20% compressed position; axle at F.S. 275.33, W.L. 36.8. The gear is to be in the forward or conventional landing position.

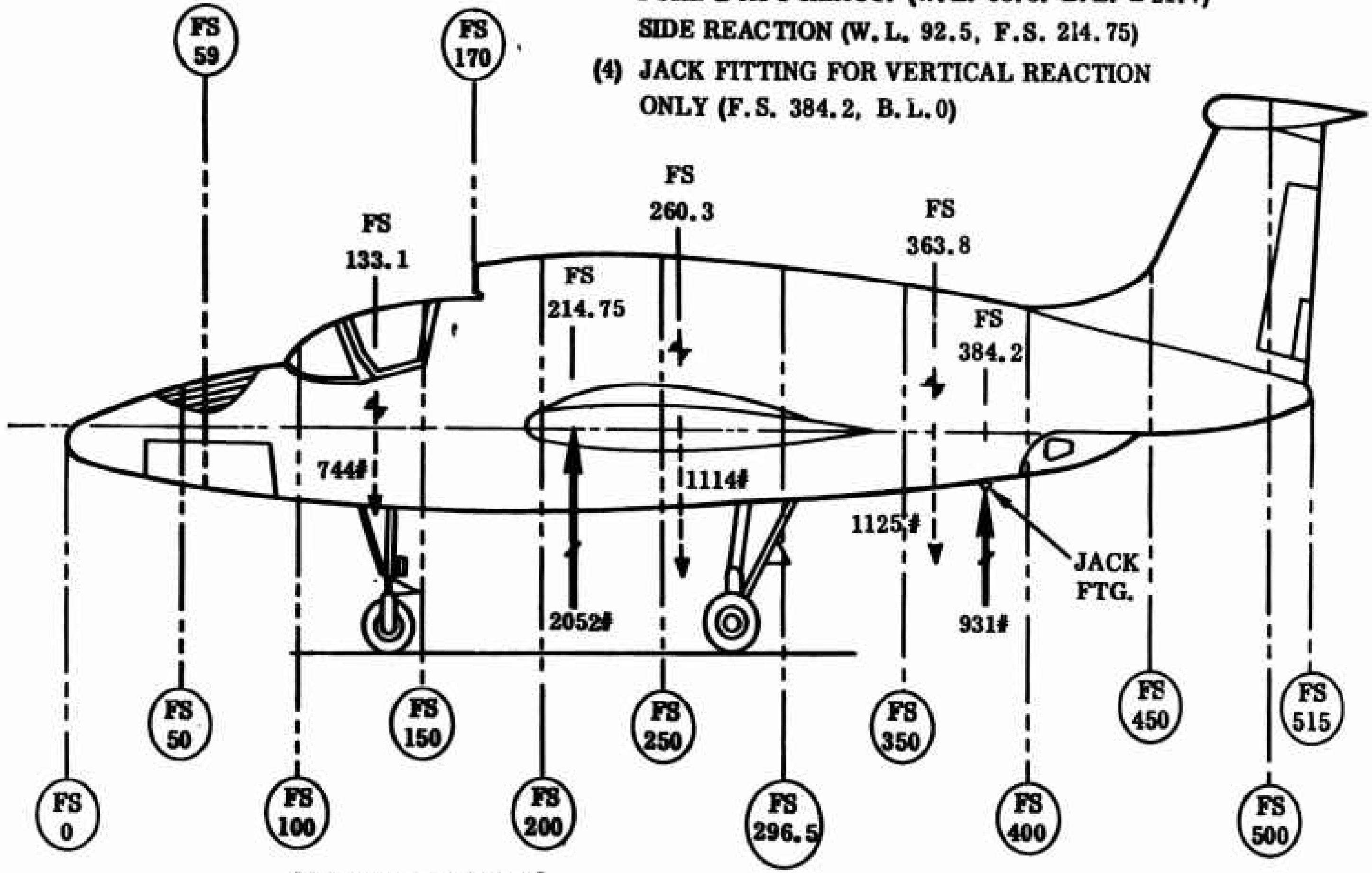
Measurements

Record readings of forward, center and aft fuselage strain gages.

Read the following deflections:

(1) Vertical deflection F.S. 00
(2) Vertical deflection F.S. 91
(3) Vertical deflection of L.H. main gear axle.
(4) Forward deflection of L.H. main gear axle.

(1) GEAR EXTENDED, CONVENTIONAL POSITION
(2) FUSELAGE REF. LINE LEVEL
(3) FRONT SPAR STATIC TEST POINTS CAPABLE OF:
VERTICAL REACTION (F.S. 214.75, B.L. ± 22.0)
FORE & AFT REACT. (W.L. 95.5. B.L. ± 21.7)
SIDE REACTION (W.L. 92.5, F.S. 214.75)
(4) JACK FITTING FOR VERTICAL REACTION ONLY (F.S. 384.2, B.L. 0)



M 214.75 = -61000 "#
M 296.5 = + 5900 "#

+ M_y = COMP. IN TOP

Figure 2 Estimated Test Article Weight Distribution for Support at Front Spar Static Test Points and Fuselage Jack Fitting

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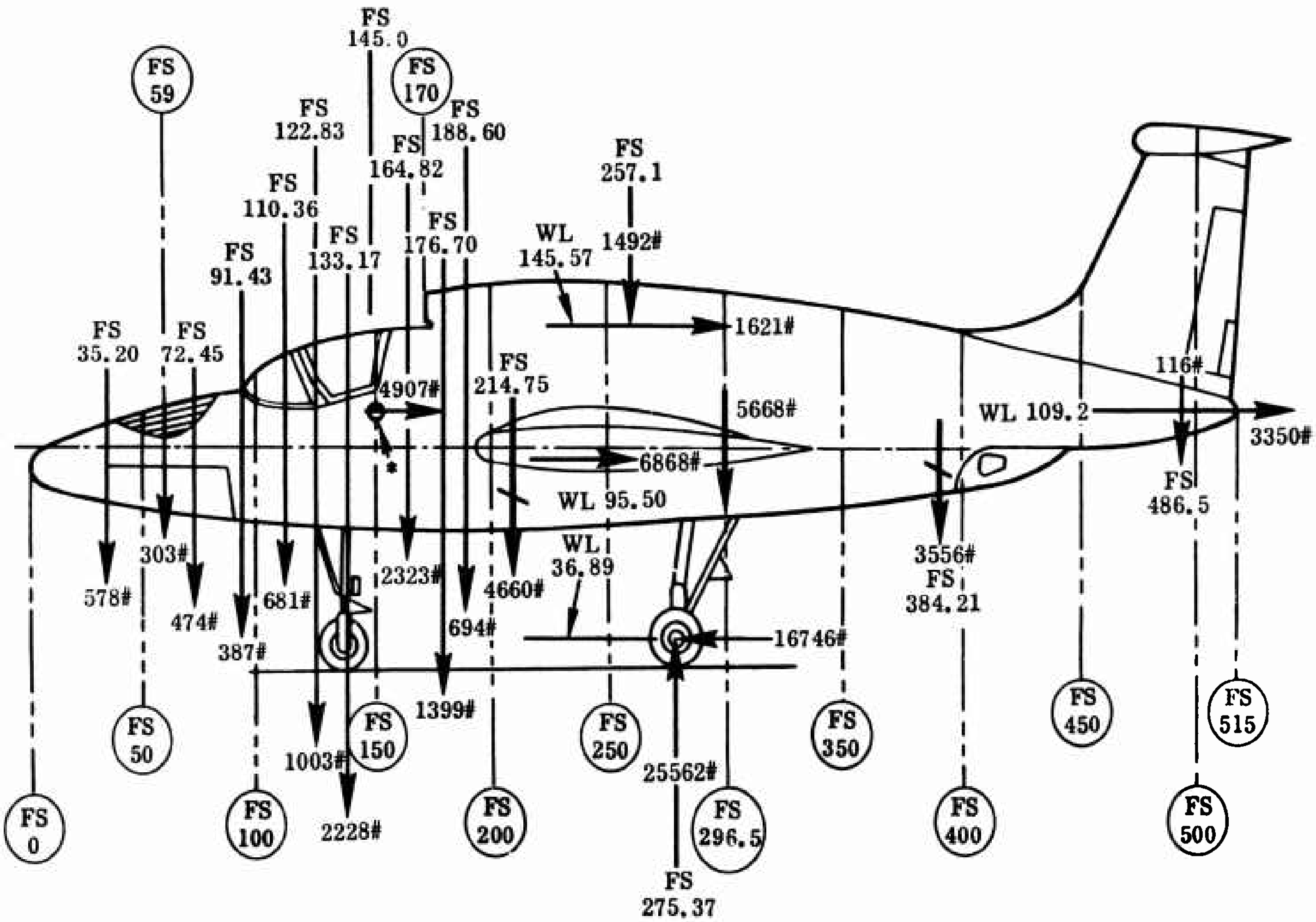


* APPLY 4907# AFT TO 143F0006
DIVIDE LOAD BETWEEN 2-143F142-1
FITTINGS @ FS - 145.00
W.L. - 111.40
B.L. - 13.03

NOTE: REACTIONS SHOWN DO NOT INCLUDE THE EFFECT OF TEST ART. W'T (SEE FIGURE 2).

Figure 3 Test No. 2, Limit Loads - 2-wheel, Tail Down, Dynamic Spring Back

TEST NO. 3

Condition

Drift Landing

Critical Structure

This is a critical condition for the main landing gear, local support structure, and fuselage center section.

Test Procedure

The airplane shall be mounted as shown in Figure 2. The reaction at the jacking fitting shall be measured, with ± 10% accuracy.

Increments of limit load as specified in the general test procedure shall be applied. Main gear vertical and forward loads shall be applied at the axles and the side loads to dummy wheels simulating ground contact point. The main gear shall be in the conventional landing position with oleos fixed in the 20% compressed position. The side load at F.S. 134.79 and W.L. 54.42 shall be applied to the nose gear with a collar to the piston below the oleo.

The side reaction at F.S. 214.75 and W.L. 95.50 is on the left hand side of fuselage. See Figure 4.

Vertical and side forces are to be applied at the rear spar-fuselage fittings; the side load coming in on the right hand side.

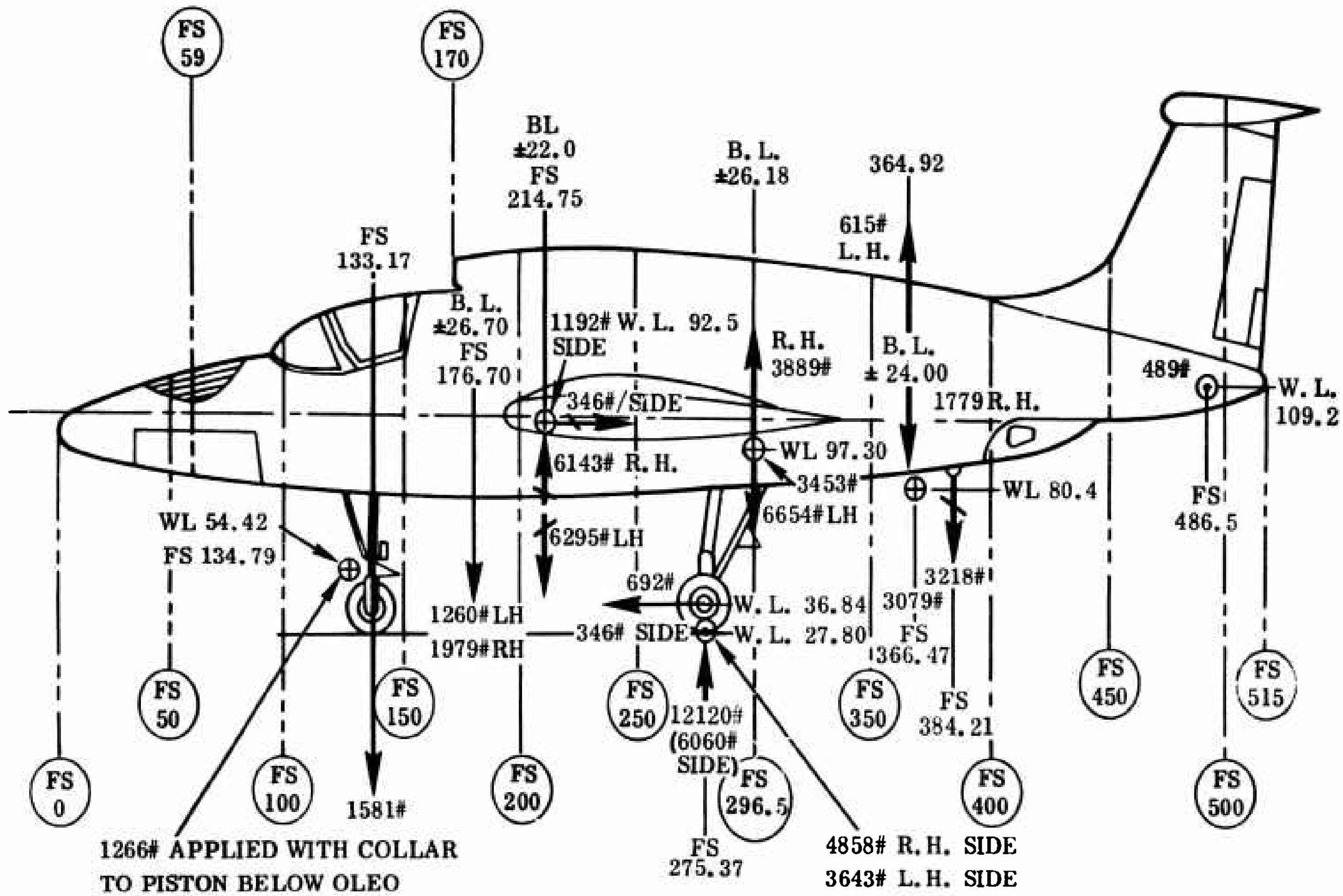
As noted, there are six reactions for this test, and care should be taken to prevent any over-constraint, so that the reactions will remain purely determinate.

Measurements

Record fuselage center section strain gages.

Read the following deflections:

(1) Vertical deflection right main gear axle .
(2) Lateral deflection right main gear axle.



⊕ LOADS PILOT'S RIGHT
⊙ LOADS PILOT'S LEFT

NOTE: REACTIONS SHOWN DO NOT INCLUDE THE EFFECT OF TEST ART. W'T (SEE FIGURE 2).

Figure 4 Test No. 3, Limit Loads - Drift Landing

TEST NO. 4

Condition

This test simulates the loading upon the structure resulting from four design symmetric-flight conditions.

(1) A high-speed 4.0 g symmetrical flight condition produces critical bending moment in the fuselage between Stations 160.0 and 250.0.
(2) A minus one g high-speed condition is critical in bending between Fuselage Stations 50.0 and 160.0 and between Stations 250.0 and 410.0.
(3) A minus two g high-speed symmetrical flight condition subjects the fuselage structure to the peak bending moments between Stations 0.0 and 50.0 and from Station 410.0 aft. This condition also produces the maximum horizontal-tail load of -7100 pounds.
(4) A 4.0 g low-speed symmetrical flight condition results in critical horizontal-tail torsion. This test simulates this critical value.

Critical Structure

This test will subject the entire fuselage and horizontal-tail structure to the design symmetrical flight loads.

Test Procedure

The test article, consisting of the complete airframe structure shall be supported as shown in Figure 1.

The vertical loads to be applied to the structure and their locations are shown in Figure 5. All fuselage vertical loads use existing strap or fitting locations.

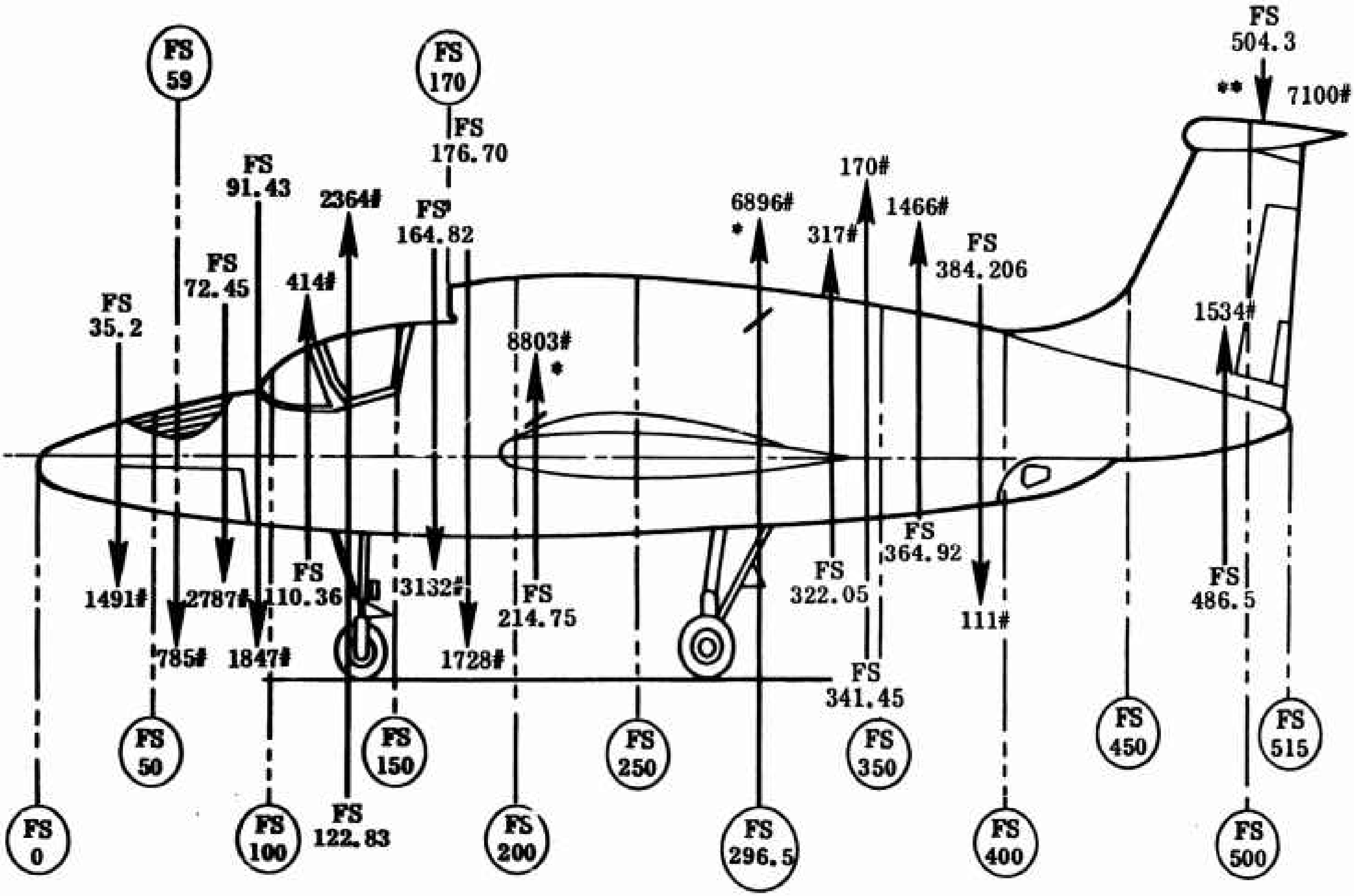
The horizontal-tail load of 7100 pounds is to be applied symmetrically to both semi-spans of the surface. The pad locations and loads of Figure 6 are to be used. The elevator shall be checked for freedom of rotation while the horizontal tail is at 100% limit load.

Measurements

All fuselage, vertical tail, and horizontal tail strain gage readings shall be recorded during the test.

Deflection gages shall be positioned at the following locations to measure deflections in the indicated directions:

Fus. Sta.	B.L.	Deflection Direction
0	0	Vertical deflection of the fuselage
35	0	"
91	0	"
150	0	"
214.75	0	"
296.5	0	"
350	0	"
400	0	"
450	0	"
500	0	"
496.08	0.0	Vertical deflection of horizontal tail center spar
496.08	+ 35.0	Vertical deflection of H.T. center spar, port mid-semispan
496.08	± 70.0	Vertical deflection of H.T. centerspar, port and stbd. tip
513.58	0.0	Vertical deflection of horizontal tail aft spar
514.32	+ 35.0	Vertical deflection of H.T. aft spar, port mid-semispan
515.06	+ 70.0	Vertical deflection of H.T. aft. spar, port tip.



*SUPPORT REACTIONS INCLUDE THE FORCES REQUIRED TO SUPPORT THE TEST ARTICLE WEIGHT: SEE FIGURE 2

**HORIZONTAL TAIL LOAD OF 7100 LBS. TO BE DISTRIBUTED IN ACCORDANCE WITH THE PAD LOADS.

Figure 5 Test No. 4, Symmetrical Flight Conditions Envelope

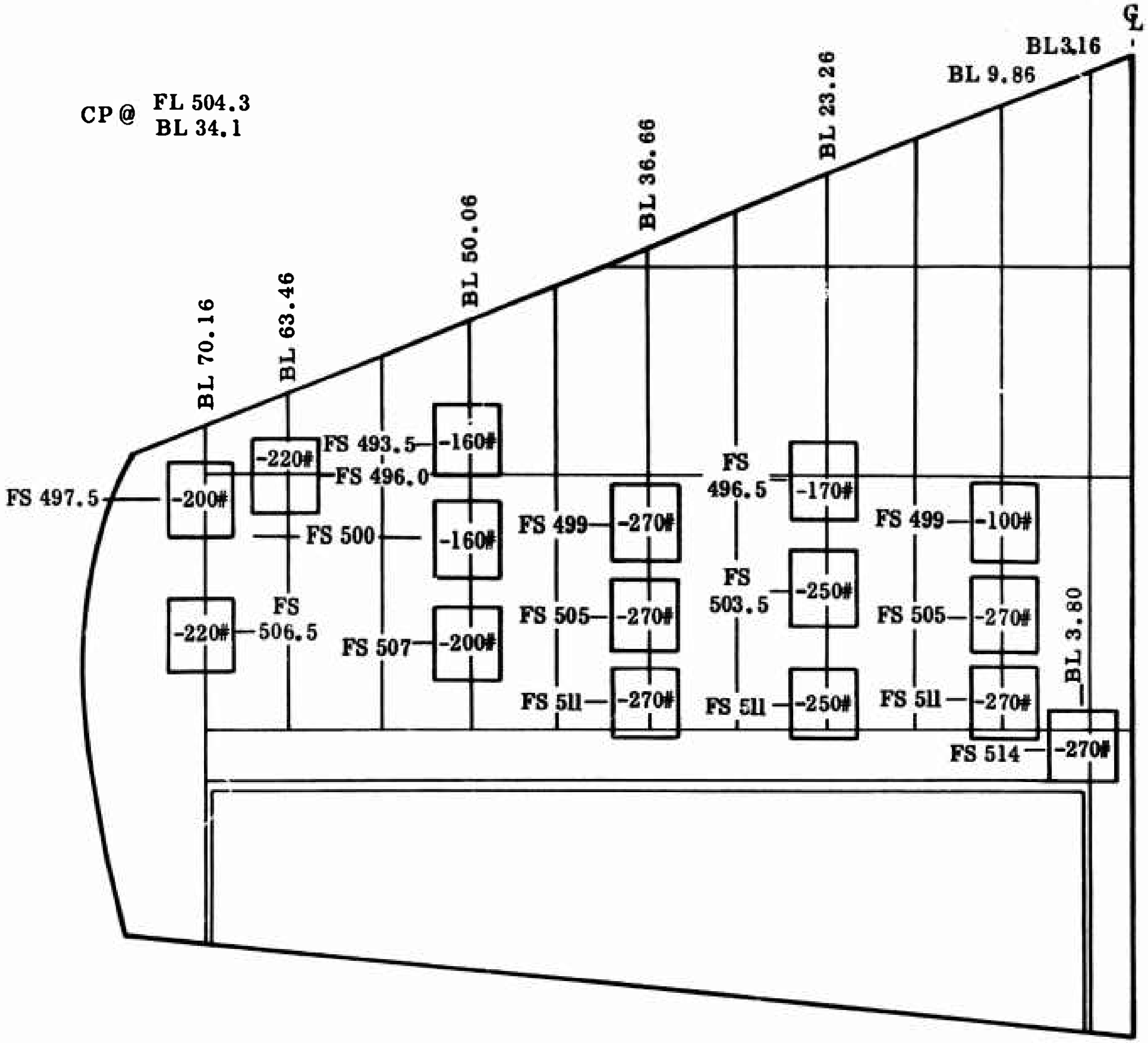


Figure 6 Limit Load Distribution for Static Test of Horizontal Tail to Simulate Maximum Shear, Maximum Bending Moment and Maximum Torque

TEST NO. 5

Condition

This test simulates the loading upon the structure resulting from the design dynamic-overswing sideslip condition. This condition subjects the structure to both vertical and lateral aerodynamic and inertia loads.

Critical Structure

The Dynamic Overswing Sideslip Condition produces critical bending and shear in the forward fuselage and vertical tail, critical torsion in the aft portion of the fuselage, and critical asymmetrical loading on the horizontal tail.

Test Procedure

The test article, consisting of the complete airframe structure shall be supported as shown in Figure 2.

The vertical loads to be applied to the structure and their locations are shown in Figure 7. All vertical loads use exsiting strap or fitting locations except for the horizontal tail load. The horizontal tail load of 1930 pounds is to be applied to the L.H. side through use of pads and whiffletrees, the same loading fixtures as designed for the maximum horizontal tail load test. A differential loading is to be applied to the straps at F.S. 133.17 to introduce a torsional loading in the fuselage.

The lateral loads to be applied to the structure are shown in Figures 8 and 9. The four loading points between Stations 35.2 and 91.0 may utilize tension pads for application of the loads. The load at Station 134.79 is to be applied to the nose gear by means of a collar to the piston below the oleo. The load at F.S.10.0 should be applied to a dummy nose cone if the actual nose cone is not available.

The vertical tail loads shown in Figures 9 and 10 shall be applied by means of tension pads at the locations indicated. The rudder shall be checked for freedom of rotation while the vertical tail is at limit load.

Measurements

All fuselage, vertical tail, and horizontal tail strain gage readings shall be recorded during the test.

Deflection gages shall be positioned at the following locations to measure deflections in the indicated directions:

FUS. STA.	W.L.	DEFLECTION DIRECTION
0	100	Vertical and Lateral Deflection of Fuselage
35	"	" " " "
91	"	" " " "
150	"	" " " "
300	"	" " " "
500	"	" " " "
500		Lateral Deflection of Vertical Tail at W.L. 200
500		Vertical Deflection of Each Horizontal Tail Tip at B.L. ± 75

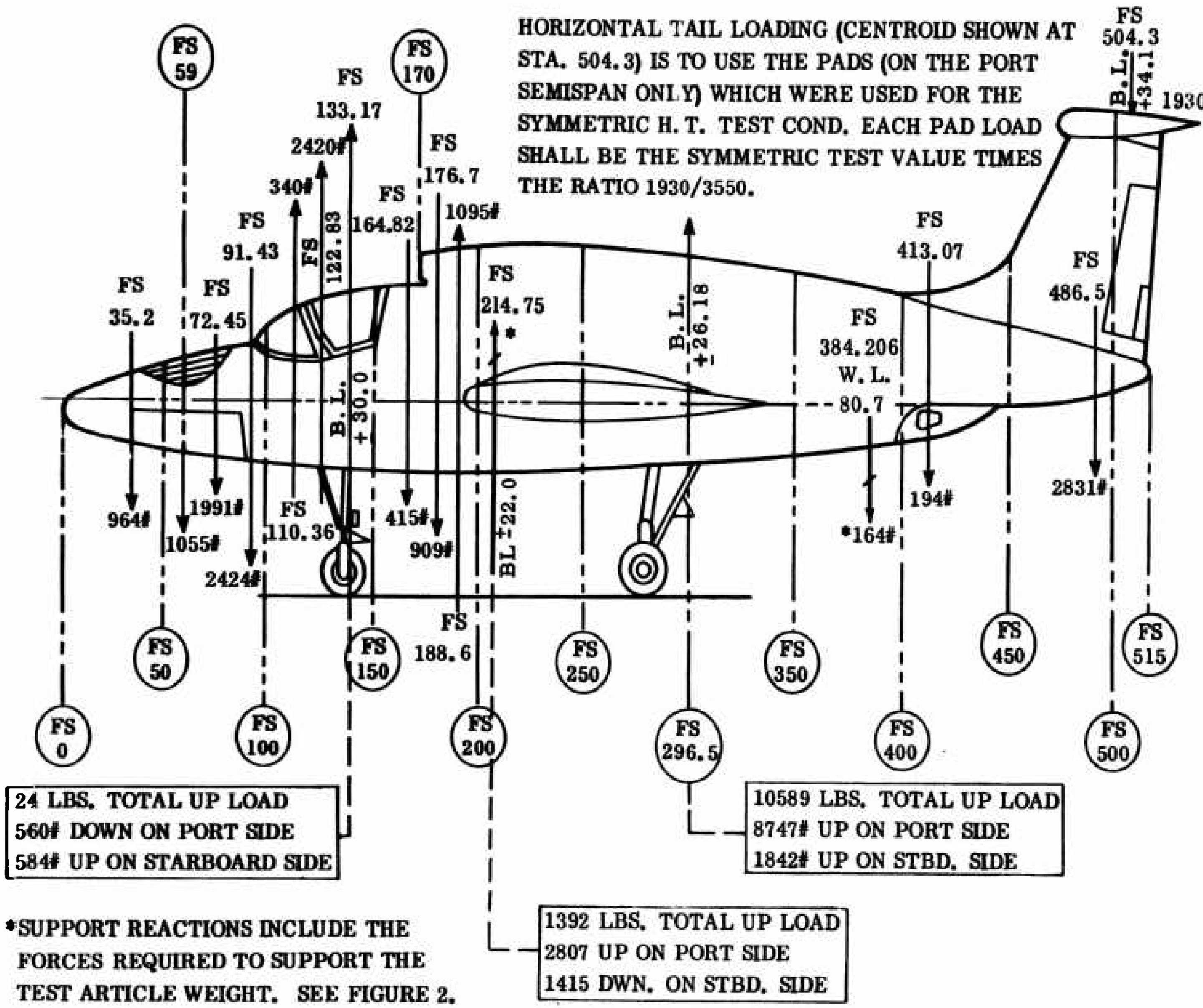
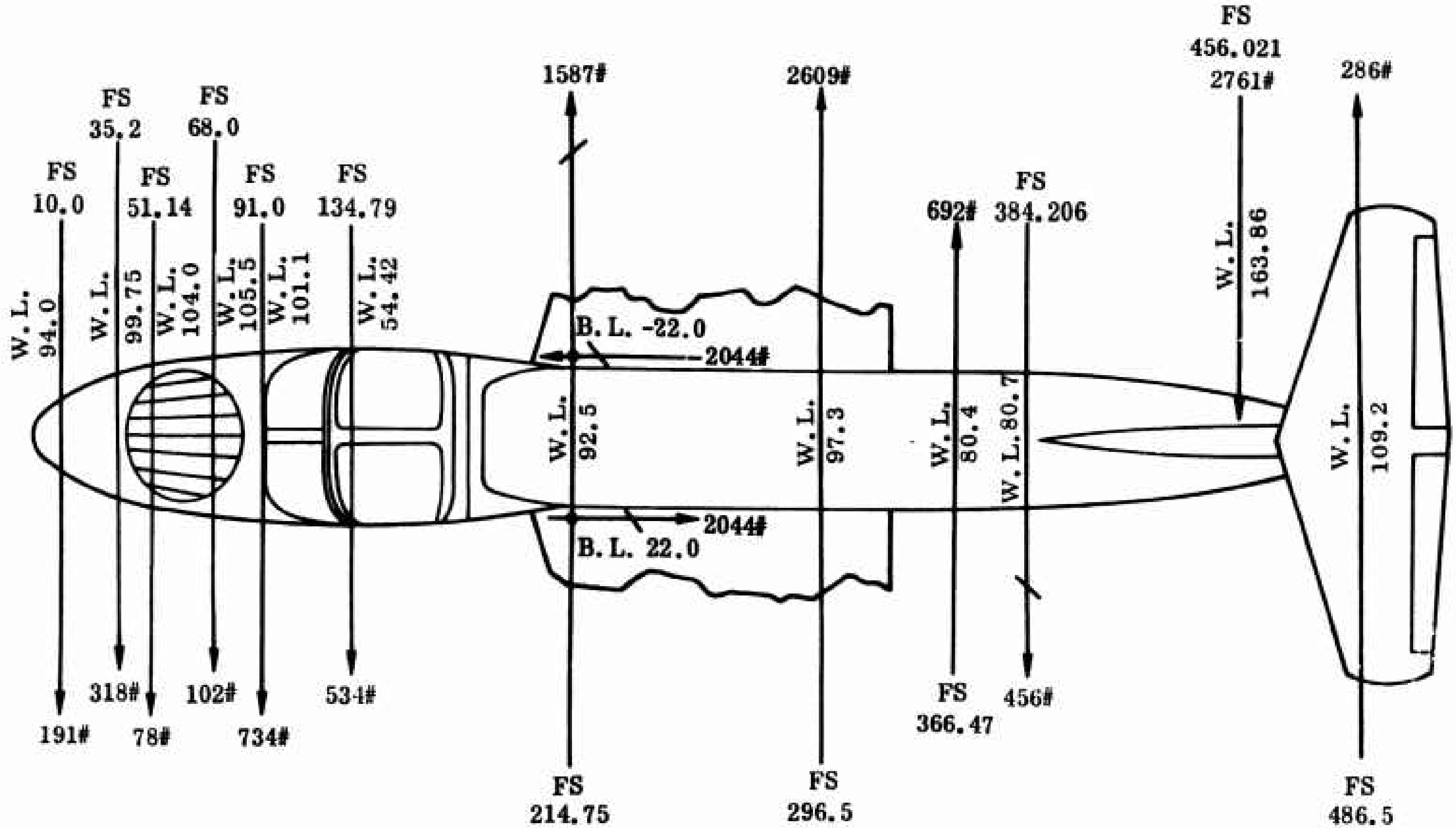
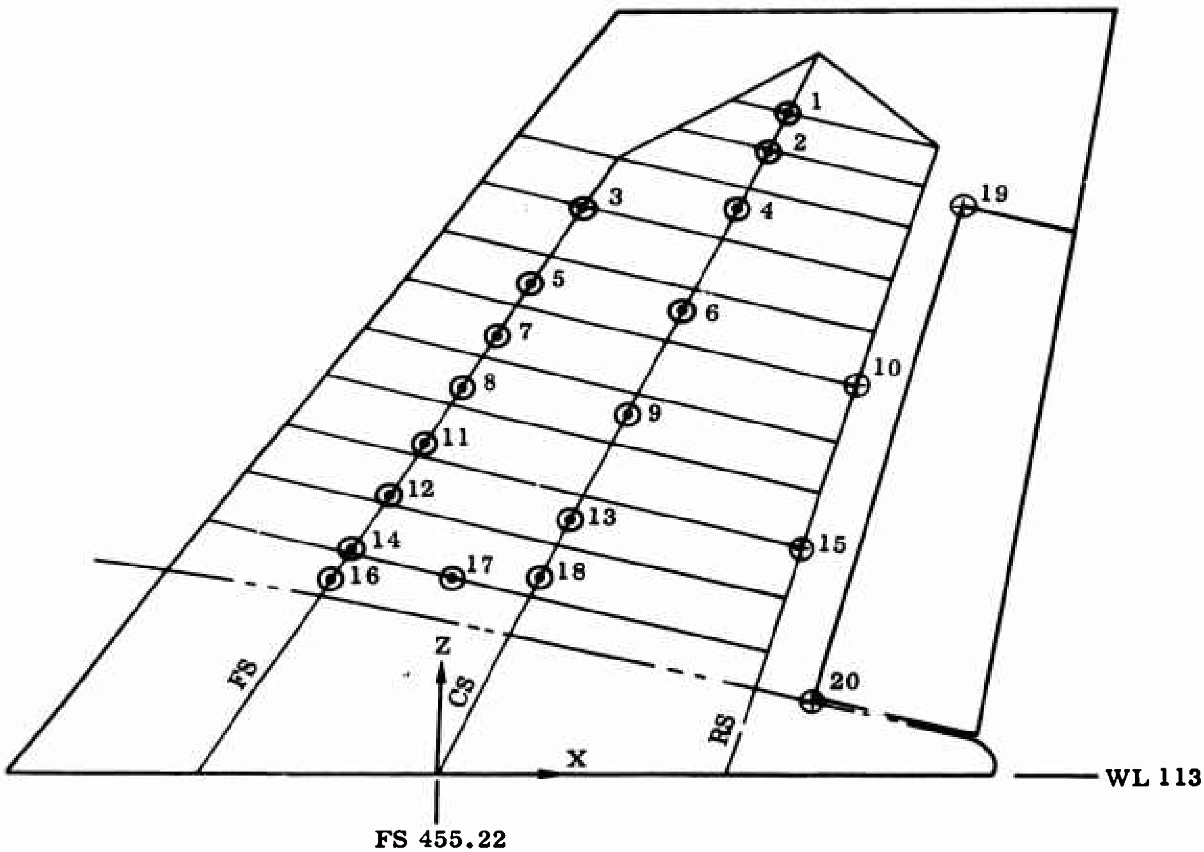


Figure 7 Test No. 5, Dynamic Overswing Sideslip Condition - Vertical Loading



VERTICAL TAIL LOADING (CENTROID SHOWN AT STA. 456.021) IS TO UTILIZE THE PAD LOADS IN THE ACCOMPANYING FIGURE.

Figure 8 Test No. 5, Dynamic Overswing Sideslip Condition - Lateral Loading



LOADS 10, 15, 19 & 20 ACT TO THE RIGHT.
ALL OTHER LOADS ACT TO THE LEFT.
LOADS 19 & 20 ARE APPLIED TO HINGE FITTINGS.

Figure 9 Vertical Stabilizer Proof Test - Distribution of Tension Pads to Simulate Maximum Loads

NO.	F (LB)	X (IN.)	Z (IN.)
1	353	37.8	80.5
2	161	35.7	76
3	279	15.8	69
4	149	32.5	69
5	236	10.3	60
6	200	26.7	56.8
7	238	6.4	53.6
8	245	2.5	47.1
9	210	20.6	43.8
10	-122	45.7	47.1
11	260	-1.5	40.5
12	268	-5.4	34.1
13	199	14.5	30.9
14	212	-9.3	27.6
15	-245	39.7	27.6
16	227	-11.5	24
17	227	1.2	24
18	156	11.3	24
19	-229	57.3	69.3
20	-263	40.2	8.6
Σ	2761	$\overline{X} = .74$	$Z = 52.7$

Positive Loads Act to the Left
Negative Loads Act to the Right
Loads #19 & #20 Are Applied to Hinge Fittings

Figure 10 Vertical Stabilizer Proof Test - Tabulation of Limit Pad Locations

TEST NO. 6

Condition

3-Point Spring Back, C.G.240, W = 9200 lb. (Nose Gear)

Critical Structure

This is a critical condition for the nose gear and its local support structure.

Test Procedure

The airplane shall be mounted as shown in Figure 1.

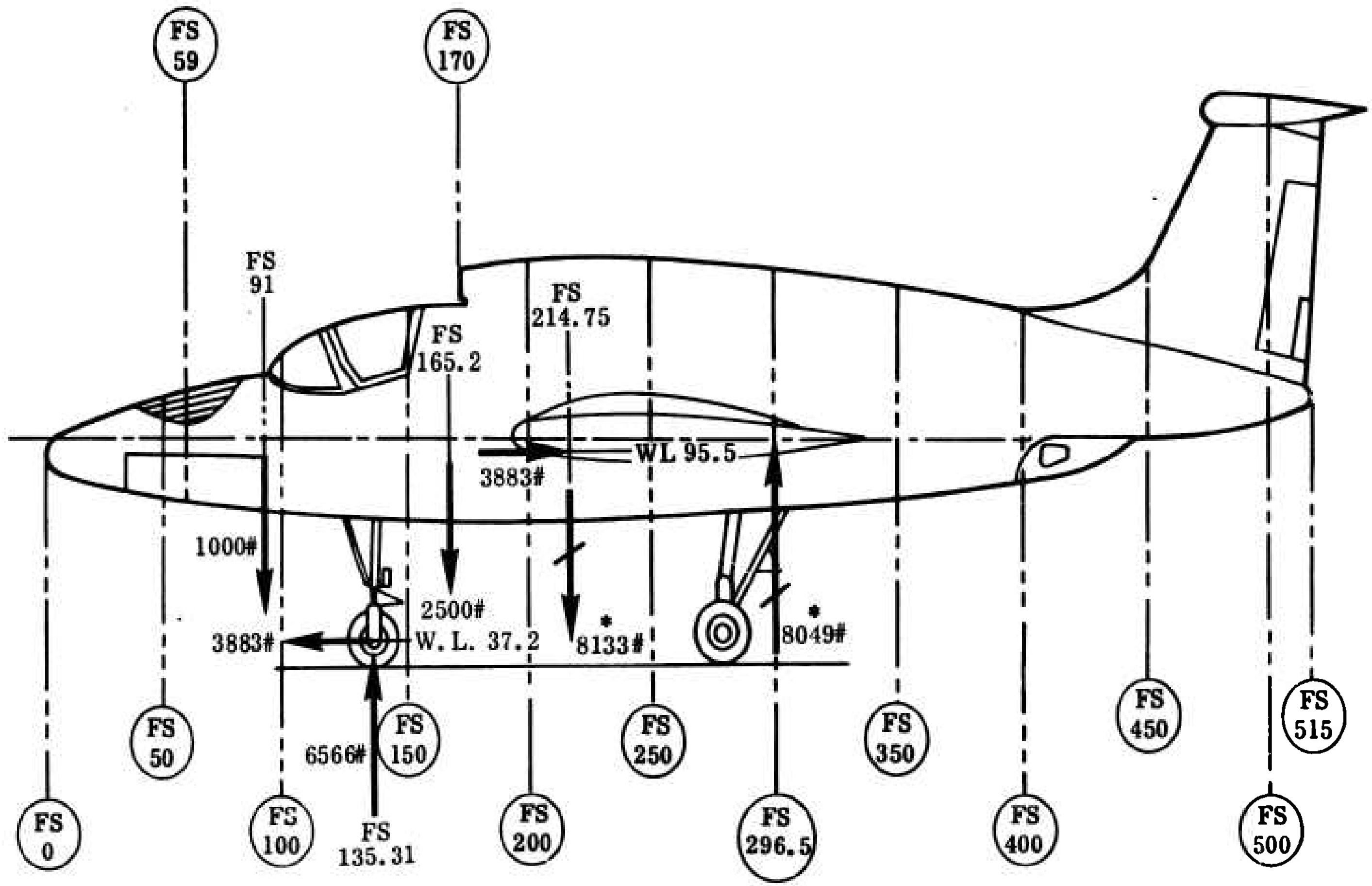
The nose gear oleo shall be fixed in the 20% compressed position and vertical and forward loads applied to the axle as indicated in Figure 11. Increments of limit load as specified in the general test procedure shall be applied.

Measurements

Record forward fuselage strain gage readings.

Read the following deflections:

(1) Vertical def. of nose gear axle.



*REACTIONS INCLUDE EFFECT OF TEST ARTICLE WEIGHT, SEE FIGURE 1

Figure 11 Test No. 6, Limit Loads - 3 point Spring Back, C.G. 240, 9200 Lb. (Nose Gear)

TEST NO. 7

Condition

Ground Turning, Gear Forward, W = 12500 lb., C.G. 240 (Nose Gear).

Critical Structure

This is a critical condition for the nose gear and its local support structure.

Test Procedure

The airplane shall be mounted as shown in Figure 1.

The nose gear oleo shall be fixed in the 20% compressed position and vertical and side loads applied to a dummy wheel at the points indicated in Figure 12. Increments of limit load as specified in the ground test procedure shall be applied.

Measurements

Record forward fuselage strain gages.

Read the following deflections:

(1) Vertical def. of nose gear axle
(2) Side def. of nose gear axle.

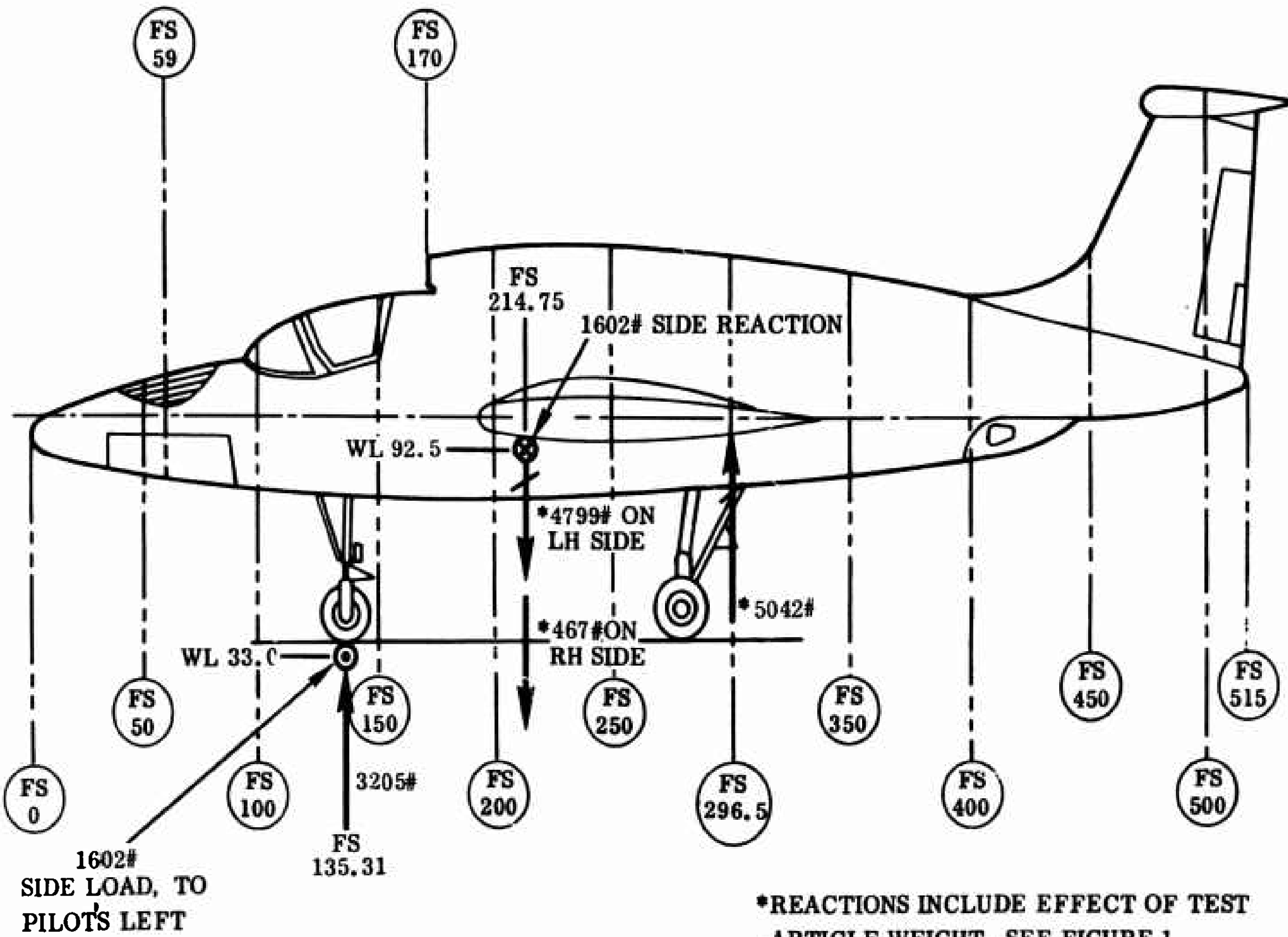


Figure 12 Test No. 7, Limit Loads - Ground Turning, Gear Fwd., 12,500 Lb. C.G. 240 (Nose Gear)

TEST NO. 8

Condition

Wing Fan Forward Support, Transition Flight, Pitching, $\beta_v = 40^\circ$ Vectored Thrust.

Critical Structure

This is a critical condition for the wing fan forward fitting; its attachment to the spar, and the inboard leading edge of the wing.

Test Procedure

The airplane shall be supported as shown in Figure 1.

Increments of limit load as specified in the general test procedure shall be applied to the wing fan forward support fitting, left hand side. Limit loads on this fitting are:

6852 lb. Forward
3581 lb. Up
2262 lb. Outboard

Calculated reactions at the spar-fuselage mounting points, including test article weight, are as follows:

L.H. Fwd. Location: 4787 lb. Dn., 13039 lb. Aft, 2262 lb. Inbd.
R.H. Fwd. Location: 3930 lb. Up, 6187 lb. Fwd.
Aft Locations: 129 lb. Up, either side

Measurements

Record readings of all L.H. wing strain gages inboard of B.L. 62.

Record the following deflections:

(1) Fwd. deflection of wing fan fwd. fitting.
(2) Up deflection of wing fan fwd. fitting.

TEST NO. 9

Condition

Wing Fan Forward and Aft Supports - Composite Condition, Hovering Flight, Rolling, $\beta_v = 0$.

Critical Structure

This is a critical condition for the wing fan forward and aft fittings and attachment, and the inboard leading edge of the wing.

Test Procedure

The airplane shall be supported as shown in Figure 1.

Increments of limit load as specified in the general test procedure shall be applied to left hand wing fan forward and aft support fittings. Limit loads are as follows:

Fwd. Fitting 2720 lb. Fwd, 5179 lb. Up, 2061 lb. Outb'd.
Aft Fitting: 5420 lb. Up.

Calculated reactions at the spar-fuselage mounting points, including test article weight, are as follows:

L.H. Fwd. Location: 8243 lb. Down, 5241 lb. Aft, 2061 lb. Inb'd.
R.H. Fwd. Location: 5006 lb. Up, 2521 lb. Fwd.
L.H. Aft Location: 8504 lb. Down.
R.H. Aft Location: 4124 lb. Up.

Measurements

Record readings of all L.H. wing strain gages inboard of B.L. 62.

Record the following deflections:

(1) Fwd. deflection of wing fan fwd. fitting.
(2) Up deflection of wing fan fwd. fitting.
(3) Up deflection of wing fan aft fitting.

TEST NO. 10

Canopy Component Test

The canopy shall be tested in a fixture to simulate the critical flight condition, which is High Speed Flight, q = 850 ps_f, 5 degrees sideslip.

Test Procedure

The canopy shall be mounted in a fixture to simulate the latch supports, shear fittings and hinge supports that exist in the airplane. The latch support need not incorporate the entire latch mechanism, but hooks which are actual aircraft parts may be used.

The canopy is to be tested to ultimate load, using the increments as specified in the general test procedure. The load is to be supplied through tension pads bonded to the Plexiglas, and the increments of load will be in terms of ultimate.

The pad loads and locations are given in the following chart. These test loads are ultimate.

Measurements

Deflections are to be taken with dial gages, or equivalent, with a range up to 2 inches. Deflection readings shall be recorded at the test increments of ultimate load up to and including 66.6% ultimate (or limit). Beyond that load, the deflection gages may be removed, if ultimate failure could cause damage to the gages.

Deflection readings shall be taken near the following pad locations:
5, 8, 12, 15, 17, 19, 34, 36 & 38.

TABLE 8

PAD LOADS FOR CANOPY STATIC TEST

PAD NO.	LOCATION F.S.	B.L.	W.L.	APPROX. DIRECTION DEG.FWD.	DEG.LEFT	ULT. TENS. PAD LOAD #
1	132		124	0	+ 90	705
2	141		125			345
3	127	L.H.	132			765
4	136		132			435
5	146		132	0	+ 90	405
6	123	+ 24	138	+15	+ 45	750
7	130	+24.5	139	0		705
8	139	+24.5	139	0		555
9	150	+24.5	139	0	+ 45	405
10	121	+15.5		+11	+ 6	735
11	129			+3	+ 5	750
12	137		Top	0	+ 5	495
13	148		Left	0	+ 5	300
14	156	+15.5		0	+ 5	225
15	121	+ 5.5		+13	0	795
16	127			+ 5		750
17	136			0		450
18	145			0		345
19	155	+5.5		0	0	225
20	132		123	0	-90	495
21	139	Right	134			278
22	126	Side	132			465
23	136		132			330
24	145		132	0	-90	248
25	124	-24.0	138	+12	-45	495
26	132	-24.5	139	0		435
27	140	-24.5	139	0		345
28	151	-24.5	139	0	-45	120
29	121	-15.5	1	+11	-6	570
30	127		Top	+ 3	-5	570
31	136		R.H.	0	-5	315
32	145			0	-5	165
33	155	-15.5		0	-5	98
34	121			+13	0	720
35	127			+ 5		705
36	136			0		435
37	145			0		248
38	155	-5.5		0	0	188

NOTES

(1) Loads are ultimate

(2) Pads are to be 6 inch square pads bonded to canopy.

(3) Load directions shown are approx. and should be applied normal to the surface.

TEST NO. 11

Proof Tests of Conventional Mode Control Systems

11 (a) ELEVATOR CONTROL SYSTEM

Condition

Limit pilot effort load (200 lb.) applied at control stick grip.

Critical Structure

Control system components and supporting brackets.

Test Procedure

Apply longitudinal load to center of control stick grip (22.25 inches from stick pivot) using load increments as specified in the general test procedure. The system is to be tested for both fore and aft loads applied to stick. Limit load is 200 pounds applied to stick in neutral position.

Applied stick load is to be reacted at elevator by (1) removing fairings for access and attaching aft end of control rod to rigid jig structure, or by (2) transmitting control torque to stabilizer by mechanically clamping elevator root ribs to stabilizer structure.

Measurements

No strain gage readings required.

Deflection

Record deflection of stick at load application point for each load increment.

11 (b) RUDDER CONTROL SYSTEM

Condition

Limit pilot effort load (300 lb.) applied at rudder pedal.

Critical Structure

Control system components and supporting brackets.

Test Procedure

Apply longitudinal load acting forward to center of rudder control pedal (16.1 inches below pivot) using load increments as specified in the general test procedure. Limit load is 300 pounds applied to pedal in neutral position.

Applied pedal load is to be reacted at rudder by removing tail cone for access and fixing rudder tension regulator to rigid jig structure.

Measurements

No strain gage readings required.

Record deflection of pedal at load application point for each load increment.

11 (c) AILERON CONTROL SYSTEM

Condition

Limit pilot effort load (100 lb.) applied at control stick grip.

Critical Structure

Control system components and supporting brackets.

Test Procedure

Apply lateral load to center of control stick grip (26.6 inches from stick pivot) using load increments as specified in the general test procedure. The system is to be tested for both left & right loads applied to stick. Limit load is 100 pounds applied to stick in neutral position.

Applied stick load is to be reacted at ailerons by (1) replacing servo link with fixed link between bellcrank and servo-actuator forward attachment point, or by (2) attaching tab control rod to rigid jig structure.

Measurements

No strain gage readings required.

Record deflection of stick at load application point for each load increment.

TEST NO. 12

Engine Mounts

Condition

12 (a) ROLLING PULLOUT

Critical Structure

Engine mount fittings and supporting tubular structure.

Test Procedure

Install following engine support fittings on test airplane:

Fwd. Mount - 143F135, 143P018 & BRE 4458 & 4995
Side Mount - 143P019 -3
Main Mount - 143P005 -11 & -13

For both tests the airplane shall be supported, fuselage reference axis level, at the front and aft spar/fuselage test fittings as shown in Figure 1. Reactions need not be measured.

Increments of limit load as specified in the general test procedure shall be applied. Detail load information (magnitude, direction & location) is shown by Figure 13 for test 12 (a) and by Figure 14 for test 12 (b); additional load information for each test is:

Test 12 (a):
Apply vertical and longitudinal loads as shown to the center and left side main mount fittings, simultaneously apply load to the truss fitting at point 81 (Ref. 2, Dwg. #143F009, Sht. 6). This load shall be applied in line with member 81 - 58 to simulate the loading on the inboard wing scroll mount. Transverse load is also applied simultaneously to the side mount (Pt. 32) along with a load applied 5 deg. fwd. of vertical at the forward support fittings (FS - 215, 80, BL 11.50L & BL 11.50R) In addition, loads simulating the inertia of the crossover ducts are applied vertically at the inboard duct support fittings, point K (FS 251.25, BL 3.22L, WL 124.53) and point N (FS 260.75, BL 3.22R, WL 124.53).

TEST 12 (b) HOVER

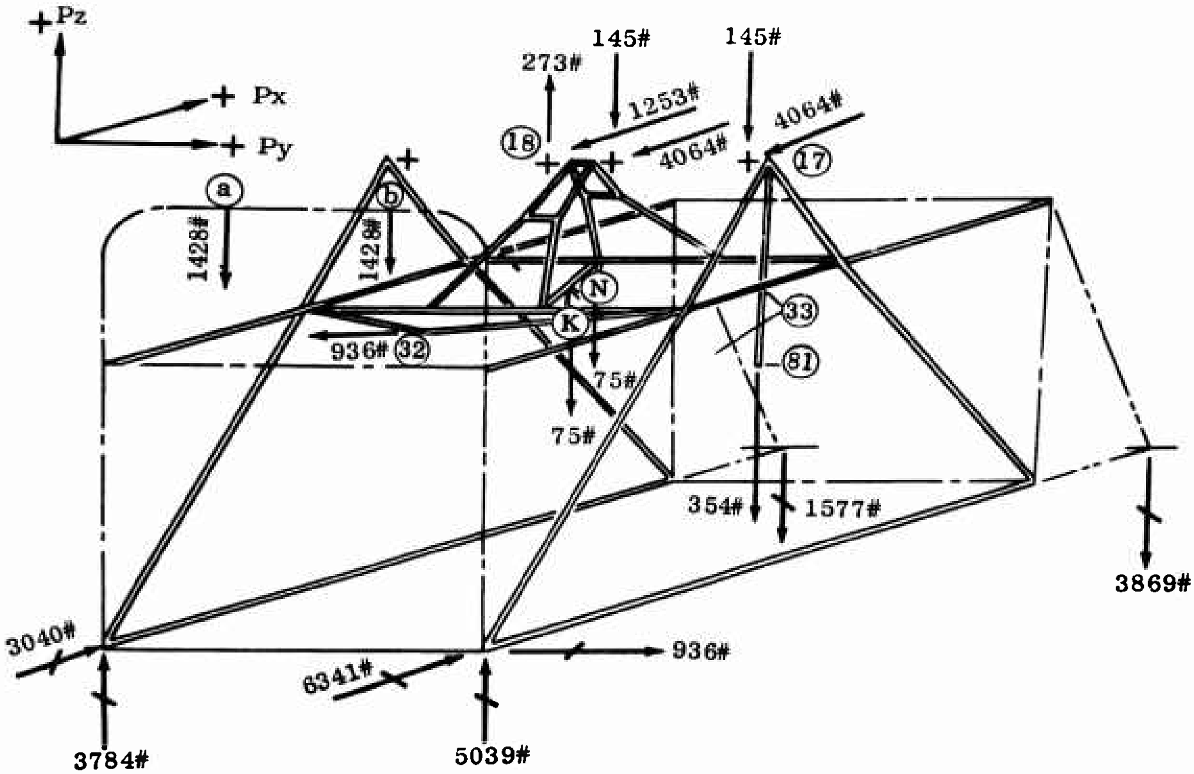
Loads are applied to engine supports similar to those of test 12 (a) except for changes in magnitude and direction. In addition, loads are applied simultaneously to the cross-over duct supports at points K and N. Thses loads shall be applied so that the following member loads result:

4-K	5150 lb. tension
10-N	5060 lb. tension
20-K	1527 lb. tension
20-N	1050 lb. tension

Measurement

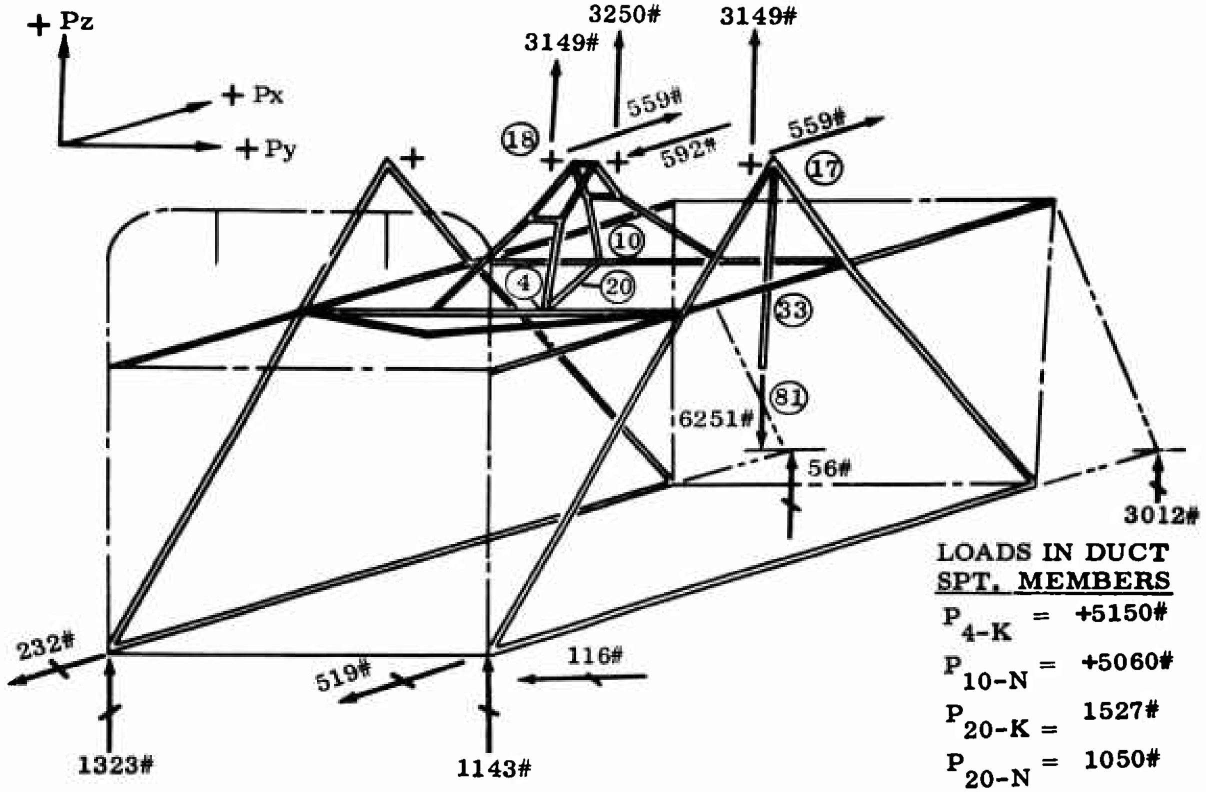
Record all 27 truss strain gages during both tests.

Record longitudinal and transverse deflections at center and left main mounts, and fwd. mount load application points.



APPLIED LIMIT LOAD	COMPONENT	LOCATION				
		ITEM	POINT	FS	BL	WL
-4064	Px	SPT. FTG.	17	257.10	20.62L	145.57
- 145	Pz	SPT. FTG.	17	257.10	20.62L	145.57
354 (DN)	PRES.	SCROLL MT.	81	256.00	22.00L	124.00
-4064	Px	SPT. FTG.	18	257.10	2.38L	145.57
-1253	Px	SPT. FTG.	18	257.10	2.38R	145.57
- 936	Py	SIDE FTG.	32	233.96	0	138.97
- 145	Pz	SPT. FTG.	18	257.10	2.38	145.57
273	Pz	SPT. FTG.	18	257.10	2.38R	145.57
- 75	Pz	DUCT FTG.	K	251.25	3.22L	124.53
- 75	Pz	DUCT FTG.	N	260.75	3.22R	124.53
-1428	Pz	FWD MT.	b	215.80	11.50L	161.34
-1428	Pz	FWD MT.	a	215.80	11.50R	161.34

Figure 13 Limit Loads - Engine Mount Test No. 12 (a)



APPLIED LIMIT LOAD	COMPONENT	LOCATION				
		ITEM	POINT	FS	BL	WL
559	Px	SPT. FTG.	17	257.10	20.62L	145.57
3149	Pz	SPT. FTG.	17	257.10	20.62L	145.57
6251 (DN)	PRES	SCROLL MT.	81	256.00	22.00L	124.00
-592	Px	SPT. FTG.	18	257.10	2.38L	145.57
559	Px	SPT. FTG.	18	257.10	2.38R	145.57
3250	Pz	SPT. FTG.	13	257.10	2.38L	145.57
3149	Pz	SPT. FTG.	18	257.10	2.38R	145.57
3588	Px	DUCT SPT.				
-1753	Py	DUCT SPT.	4	243.00	7.07L	132.00
-3249	Pz	DUCT SPT.				
-2972	Px	DUCT SPT.				
2035	Py	DUCT SPT.	10	267.00	7.45R	132.00
-3553	Pz	DUCT SPT.				
-391	Px	DUCT SPT.				
128	Py	DUCT SPT.	20	256.75	0.00	132.00
-2028	Pz	DUCT SPT.				

Figure 14 Limit Loads - Engine Mount Test No. 12 (b)

TEST NO. 13

Windshield Proof Test

The windshield shall be proof tested to limit load on the airplane to simulate the critical condition, which is High-Speed Flight, $q = 850$ psf, 5 degrees sideslip.

Test Procedure

The airplane shall be supported as shown in Figure 1. Reactions need not be measured.

The windshield shall be loaded to limit load in the increments specified in the general test procedure. Both positive and negative pressures are required in this test. If tension pads are used incorporating a bond, the effect of bond and solvent to remove it must not damage the windshield, which is a flight article. If it is found to be economical, the Plexiglas windshield may be replaced after test, provided permission of the Project Engineer is obtained. Pad loads are given in the following chart.

Deflection Readings

Deflections are to be taken with dial gages, or equivalent, with a range up to 2 inches. Readings shall be made near the following locations:
103, 105, 111, 112, 204, 205, 206, 208, & 209.

TABLE 9

PAD LOADS FOR WINDSHIELD STATIC TEST

TENSION PADS (EXTERNALLY APPLIED)

PAD NO.	LOCATION F.S.	LOCATION B.L.	LOCATION W.L.	PITCH DEG. FWD.	YAW DEG. LEFT	ROLL DEG. LEFT	LIMIT PAD LOAD
101	115.7	29	121.5	-	90	90	450
102	108.2	28	122.6	-	85	90	530
103	112.7	27.5	128.0	-	85	90	470
104	100.5	26.0	122.5	-	80	90	440
105	105.2	25.5	127.1	-	80	85	300
106	94.4	22	122	60	45	45	200
107	99.8	21.4	127.4	55	45	45	135
108	104.9	21.5	131.6	45	45	45	150
109	109.7	23.3	133.8	30	50	50	450
110	108.7	15	136.5	30	-3	-3	240
111	108.7	6	136.8	30	0	0	210
112	108.7	-6	136.8	30	0	0	210
113	108.7	-15	136.5	30	-2	-2	140
114	95.7	-22	122.2	60	-45	-45	260
115	101.7	-22	129.1	50	-45	-45	260
116	109.2	-22	134.5	30	-45	-45	250
117	100.5	-26	122.5	-	-85	-85	240
118	106.7	-27	124.7	-	-90	-90	280
119	112.2	-27.5	128.0	-	-90	-85	280
120	115.7	-29	121.5	-	-90	-90	260

NOTES

(1) Load directions are given with reference to pitch, yaw, and roll axes:

(2) Pad locations and vectors are approximate. Loads should be applied normal to surface.

TABLE 10

COMPRESSION PADS (EXTERNALLY APPLIED)

PAD NO.	LOCATION F.S.	LOCATION B.L.	LOCATION W.L.	PITCH DEG. FWD.	YAW DEG. LEFT	ROLL DEG. LEFT	LIMIT PAD LOAD
201	91.1	12	125	40	0	0	360
202	96.4	12	129.3	40			220
203	101.7	12	133	35			67
204	91.1	0	125	40			380
205	96.4	6	129.3	40			310
206	101.7	0	133	35			170
207	91.1	-12	125	40			470
208	96.4	-12	129.3	40			340
209	101.7	-12	133	35	0	0	250

TEST NO. 14

Main Landing Gear Door Proof Test

The main landing gear door shall be proof tested to limit load on the airplane to simulate the critical condition, which is High-Speed Flight, q = 850 psf.

Test Procedure

The airplane shall be supported, fuselage Ref. line level, at the four spar-fuselage static test locations. Reactions need not be measured.

The main landing gear should be retracted and the main landing gear doors closed and latched. Outer and inner panels of the left hand main landing gear door shall be loaded with tension pads in the increments specified in the general test procedure. Limit loads to be applied to particular areas of the door are shown in Figures 15 and 16.

Measurements

Deflection with hanging scales, or equivalent, having a range up to two inches shall be measured at the following locations, deflections relative to the fuselage at F.S.300:

(1) F.S. 300, Upper edge of outer panel.
(2) F.S. 300, Hinge line between outer & inner panels.
(3) F.S. 300, Hinge line, inner panel to fuselage.

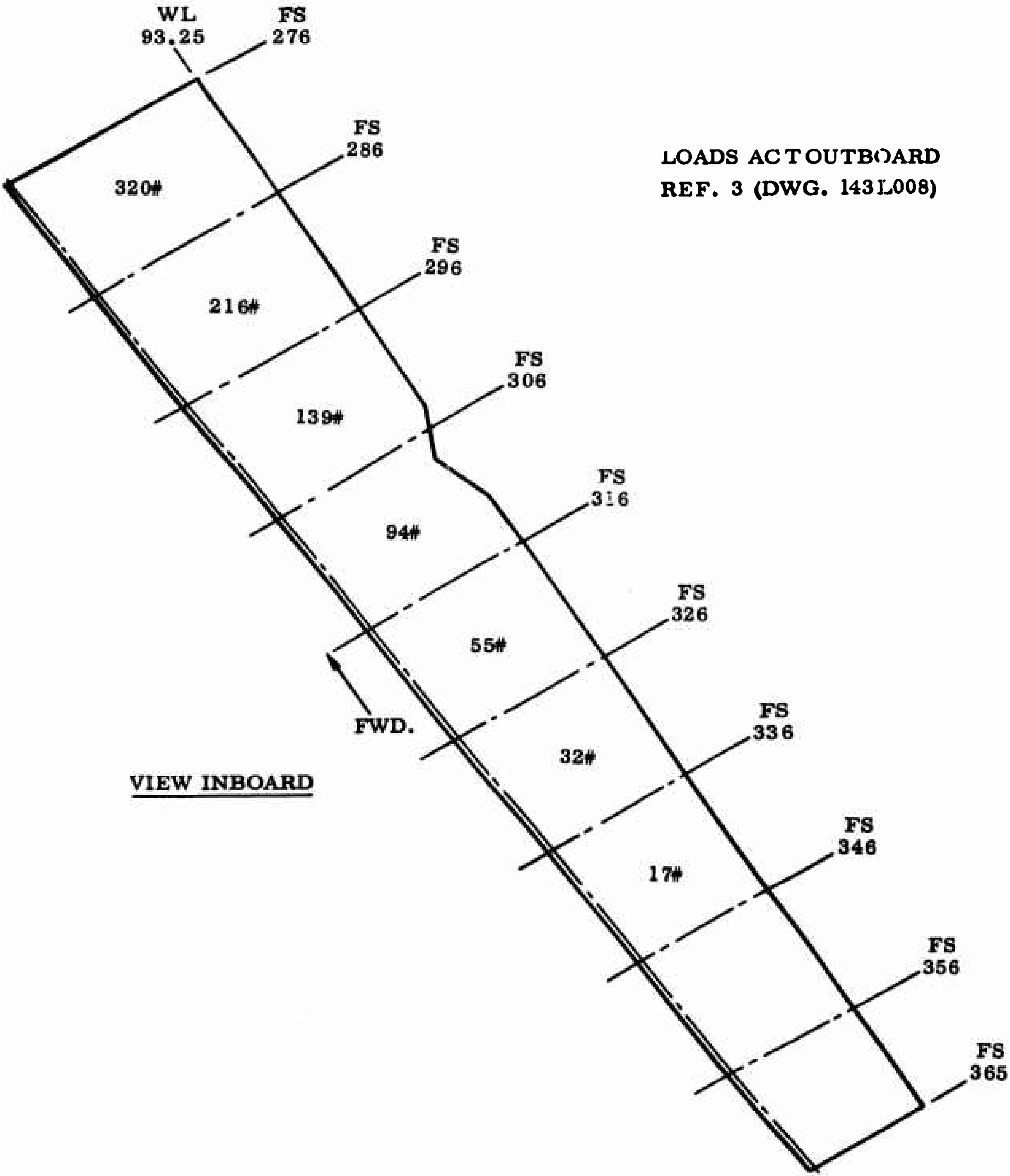


Figure 15 Main Landing Gear Door - Outer Panel, Limit Test Loads

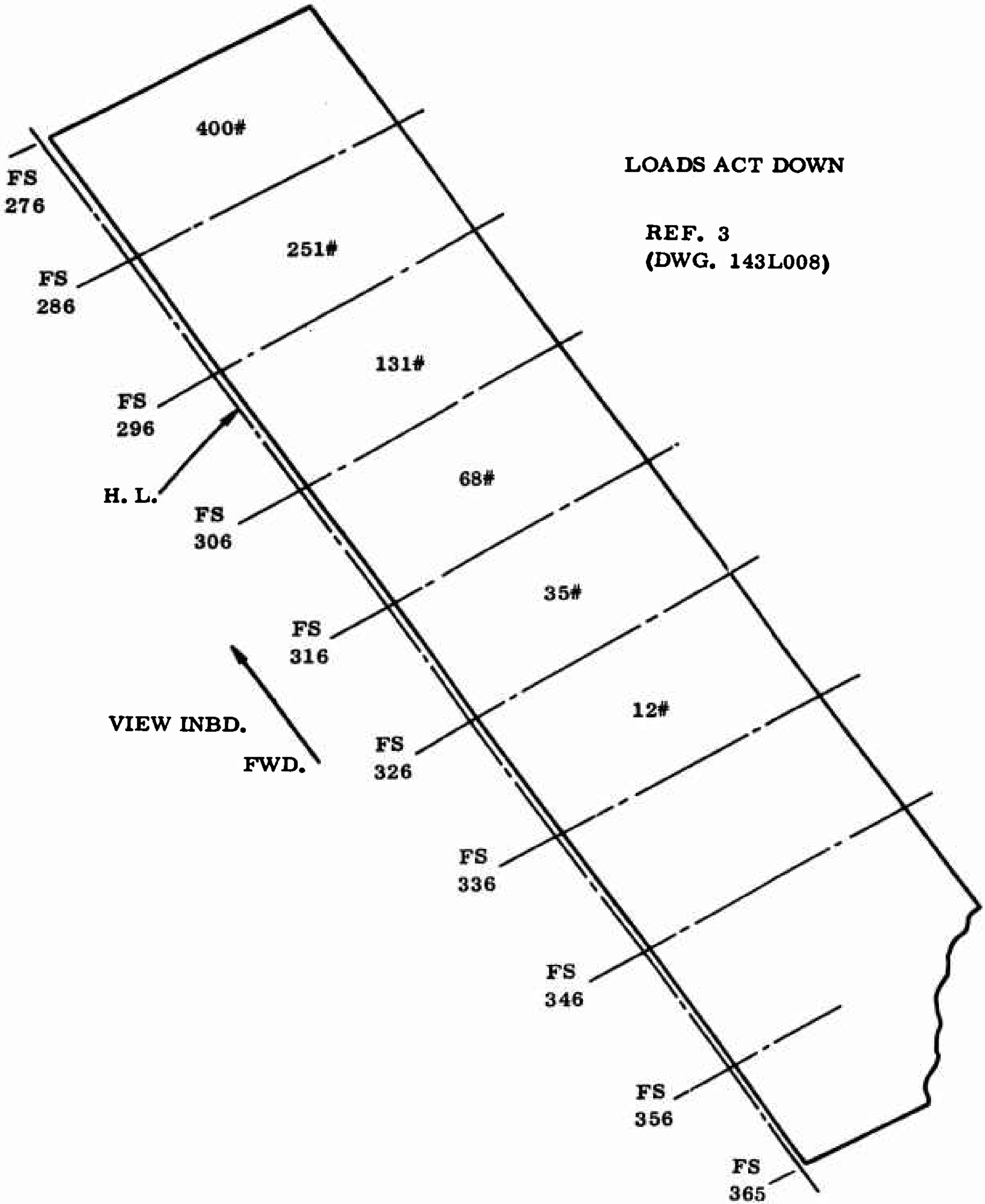


Figure 16 Main Landing Gear Door - Inner Panel, Limit Test Loads

TEST NO. 15

Flap Proof Test

The flap shall be removed from the airplane and proof tested in a fixture to simulate the critical condition, which is Flap Fully Deflected, V = 180 K.

Test Procedure

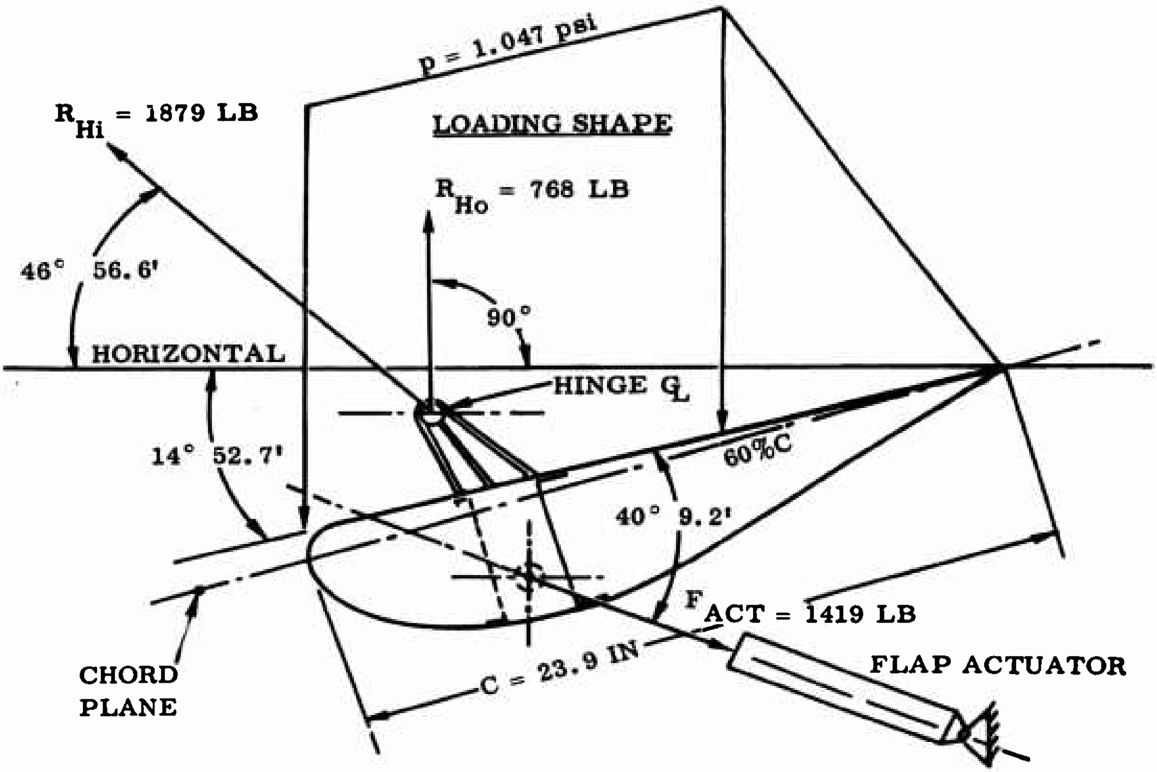
The flap shall be mounted in a test fixture to accommodate the two hinges. The link simulating the actuator shall be calibrated to measure the reacting axial load in that member. The flap shall be positioned as shown in Figures 17 and 18, and loaded to the limit load distribution shown, using lead shot or sand bags which will approximate the pressure distribution. The load increments in percent of limit shall be: 20 - 50 - 20 - 80 - 20 - 100 - 20.

Measurements

Hanging scale deflection readings shall be taken at the increments of load above, and the points to be as follows:

(1) Vertical deflection of leading edge at mid-span.
(2) Vertical deflection of trailing edge at mid-span.

RESULTANT LOAD R = 1536 LB *



R_{Hi} = INBOARD HINGE REACTION
R_{Ho} = OUTBOARD HINGE REACTION

*TOTAL FLAP LOAD NORMAL TO CHORD PLANE = 1472 # LIMIT
*TOTAL FLAP LOAD PARALLEL TO CHORD PLANE = 439#

Figure 17 Flap Test Load

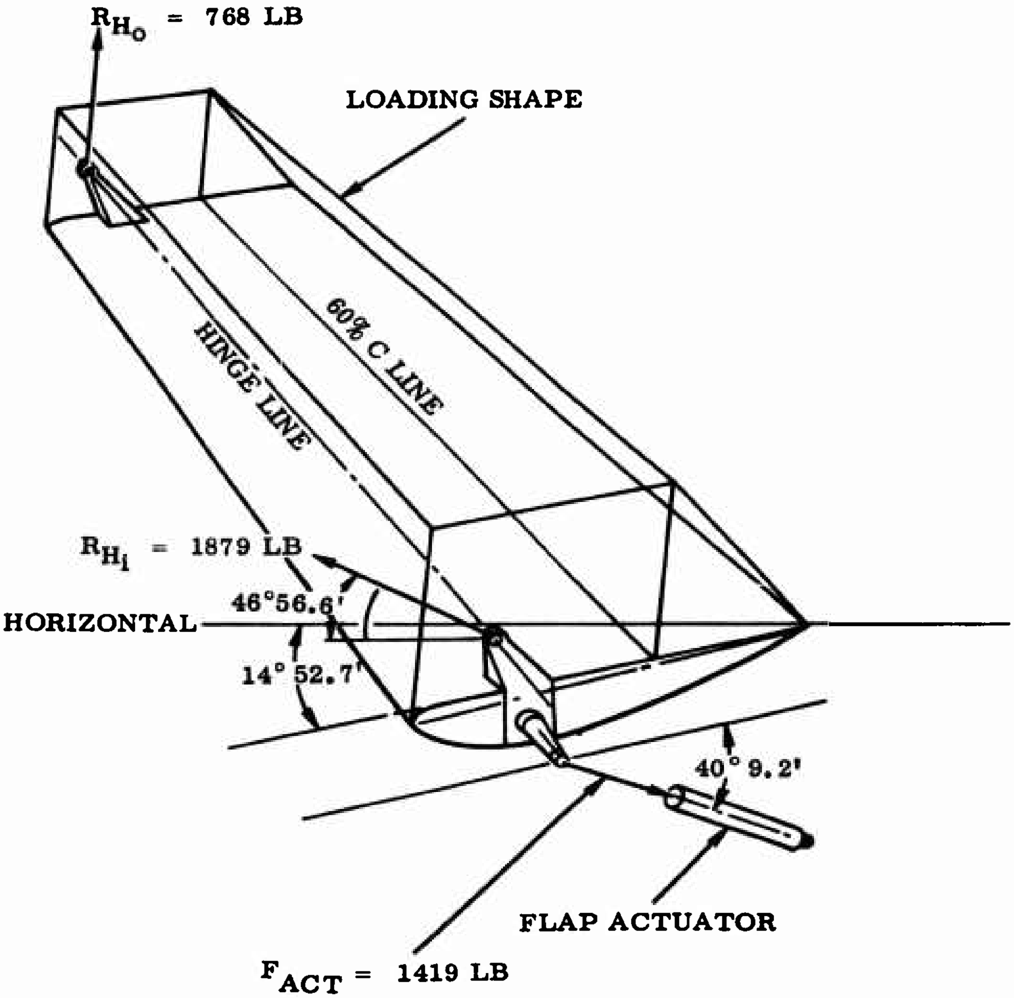


Figure 18 Flap Test Load (Limit Loads)

TEST NO. 16

Rudder Proof Test

The rudder shall be removed from the airplane and proof tested in a fixture to simulate the critical condition, which is considered to be a rudder-induced maneuver in which pilot effort load is reacted at the surface, $V = .60\ V_H$, $q = 234$ psf.

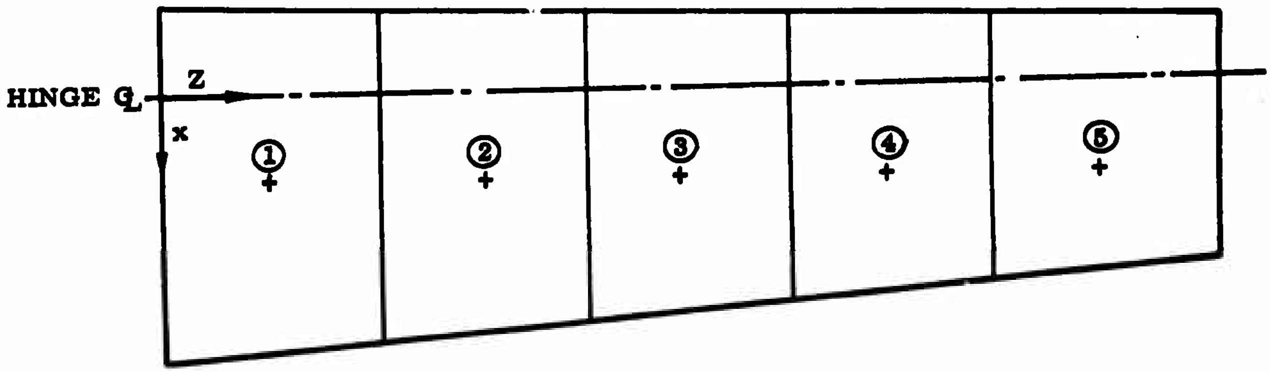
Test Procedure

The rudder shall be mounted, chord plane horizontal, in a test fixture to accommodate the two hinges and provide a torque reaction for the rudder torque tube. The rudder shall be loaded to the limit load shown in Figure 19, using lead shot or sand bags to approximate the distribution. The load increments in percent of limit load shall be as follows: 20 - 50 - 20 - 80 - 20 - 100 - 20.

Measurements

Hanging scale deflection readings shall be taken at the increments of load above and the points to be as follows:

(1) Leading edge deflection at mid-span.
(2) Trailing edge deflection at mid-span.



	W, #	x CP, INCH	Wx	Z CP, INCH
1	300	1.95	585	6.5
2	270	1.8	486	19.0
3	250	1.7	425	31.0
4	235	1.6	375	43.0
5	235	1.5	353	49.5
			2225"#	

RUDDER LIMIT LOAD = 1290#
RUDDER LIMIT HINGE MOMENT = 2220"#
(BASED ON 300# PEDAL FORCE)

Figure 19 Rudder - Estimated Limit Load Distribution

TEST NO. 17

Elevator Proof Test

The left hand elevator shall be proof tested to limit load on the airplane to simulate the critical condition, which was found to be a composite of pitching maneuvers in which pilot effort load is reacted at the surface.

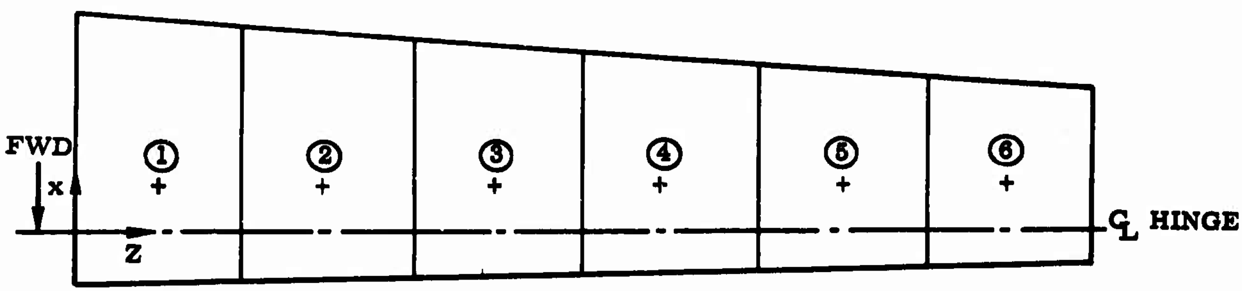
Test Procedure

The elevator shall be installed on the airplane with the elevator chord plane horizontal; the torque reaction to be furnished by disconnecting the centerline actuating link from the control system and connecting it to rigid jig structure. The elevator shall be loaded to the limit load shown in Figure 20, using lead shot or sand bags to approximate the distribution. The load increments in percent of limit load shall be as follows: 20 - 50 - 20 - 80 - 20 - 100 - 20.

Measurements

Hanging scale or dial gage readings shall be taken at the following points. The deflections should be the vertical deflections relative to the horizontal stabilizer rear spar:

(1) Elevator L.E. midway between inboard & center hinges.
(2) Elevator T.E. midway between inboard & center hinges.
(3) Elevator L.E. midway between outboard & center hinges.
(4) Elevator T.E. midway between outboard & center hinges.



	W, #	x CP, INCH	W x	Z CP, INCH.
1	125	2.4	300.0	5.5
2	115	2.3	264.5	16.5
3	110	2.2	242.0	27.5
4	100	2.1	210.0	38.5
5	95	1.9	182.2	49.5
6	90	1.9	171.0	60.5
	635		1369.7	

ELEVATOR LIMIT LOAD = 634#
ELEVATOR LIMIT HINGE MOMENT = 1370"#
(BASED ON 200# STICK FORCE)

Figure 20 Elevator - Estimated Limit Load Distribution

TEST NO. 18

Aileron Proof Test

One aileron shall be proof tested to limit load on the airplane to simulate the critical condition, which occurs at the beginning of a 500 knot, sea level roll with full aileron deflection.

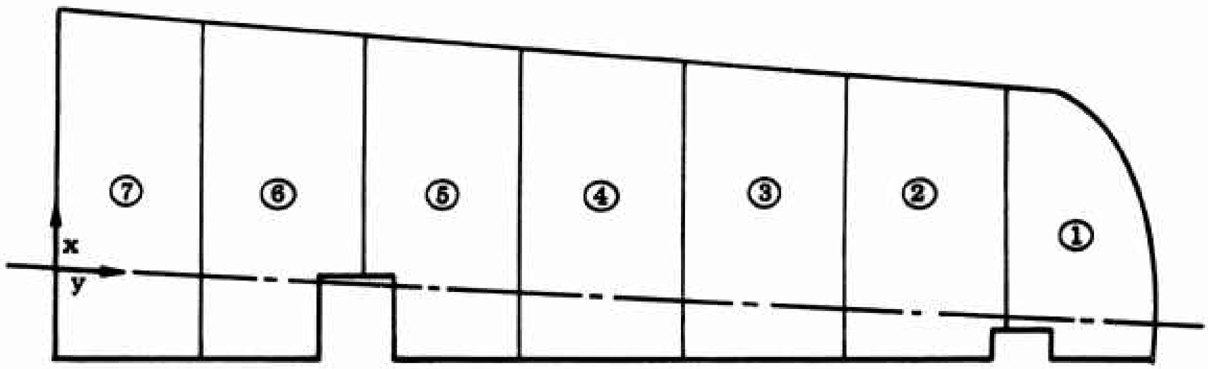
Test Procedure

The aileron shall be installed on the airplane in the neutral position and fixed rigidly at the booster to react the hinge moment. The loading shall be supplied with tension pads to approximate the distribution shown in Figure 21. The load increments in percent of limit load shall be in accordance with those given in the general test procedure.

Measurements

Hanging scale or dial gage readings or equivalent be taken at the following points; the deflections should be the vertical deflections relative to the wing rear spar:

(1) Aileron L.E. midway between inboard & center hinges.
(2) Aileron T.E. midway between inboard & center hinges.
(3) Aileron L.E. midway between outboard & center hinges.
(4) Aileron T.E. midway between outboard & center hinges.



	W, #	x CP, INCH.	W x	y, INCH.
①	230	3.0	702.0	71.3
②	440	3.2	1408.0	60.3
③	465	3.3	1534.5	49.3
④	480	3.5	1680.0	38.3
⑤	480	3.6	1728.0	27.3
⑥	485	4.0	1940.0	16.3
⑦	545	3.7	2016.5	5.3
	3125		11,009.0	

Figure 21 Aileron - Estimated Limit Load Distribution

TEST NO. 19

Nose Gear Door Proof Test

The nose gear door shall be proof tested to limit load on the airplane to simulate the critical condition, which is High-Speed Flight, V = 500 knots at Sea Level.

Test Procedure

The nose gear doors shall be installed with nose wheel removed. One of the forward doors should be disconnected from the actuating link, while the other forward door is up and locked. Lead shot bags may be used to simulate aerodynamic pressure, reaching through the open door to lay the bags on the inside of the closed forward door and also on the aft part of the door. (Ref. Dwgs. 143L014 and 143L015). The limit pressure is approximately 0.8 psi and may be distributed uniformly. The load increments in percent of limit load shall be as follows: 20 - 50 - 20 - 80 - 20 - 100 - 20.

Measurements

Hanging scale or dial gage readings shall be taken at the following points. The deflections should be the vertical deflections relative to local fuselage structure:

(1) Fwd.Door: Inboard forward corner.
(2) Fwd.Door: Inboard aft corner.
(3) Aft Door: Fwd. edge at B.L.O.

TEST NO. 20 (a)

Condition

Transition Flight, Wing Fan Door Open, Yaw Right.

Critical Structure

Wing fan door, wing fan door actuator arms, outrigger arms, support structure, General Electric fan hub, and struts.

Test Procedure

The wing fan shall be mounted vertically to a rigid jig using actual fan fittings with the two fan doors open and parallel to the ground.

Increments of limit loads shall be applied using shot bags distributed as shown in Figure 22 through 25. The door shall be supported in three ways:

1) With all 4 actuators in operation
2) With only the outboard aft and inboard forward actuators
3) With only the outboard forward and inboard aft actuators

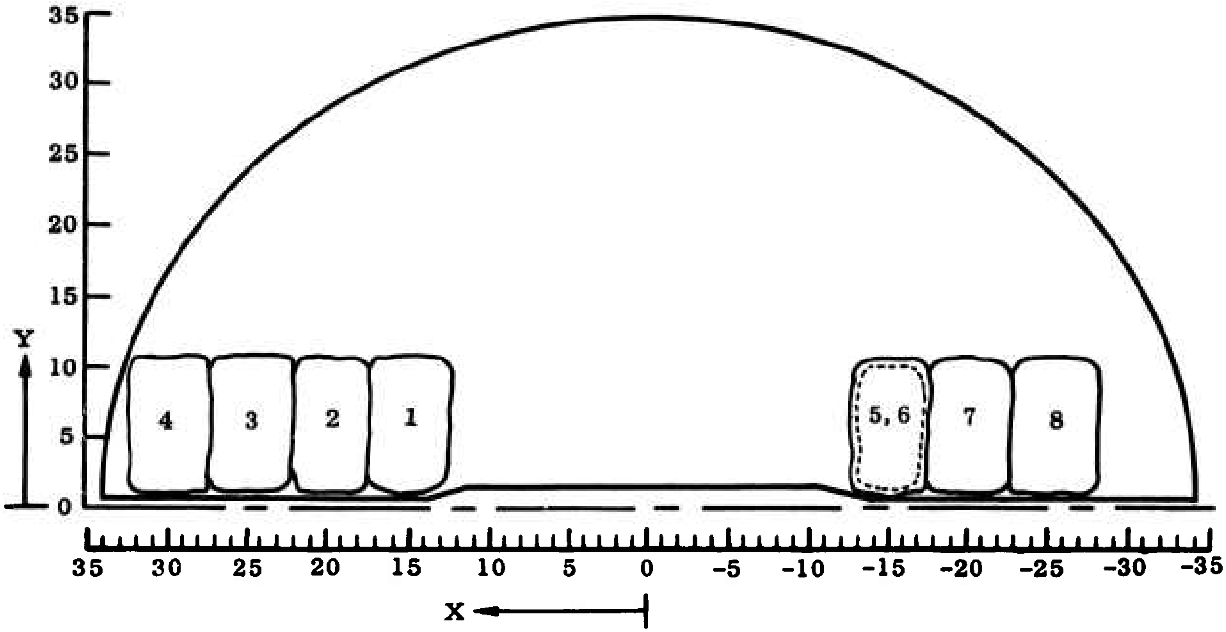
Instrumentation shall be provided to measure side reactions (airplane axis) at the forward and aft outrigger arm hinge centerline. Some other method of loading may be used, provided proper limit loads and distributions are simulated.

Measurements

Load measuring bolts shall be used to measure the load on the "record player", G.E. part number 4012001-300-2. These bolts will be used eight places, replacing NAS 1271H10 bolts, which attach 143W089 actuator bracket to G.E. 4012001-300-2 "record player" (reference print 143W042).

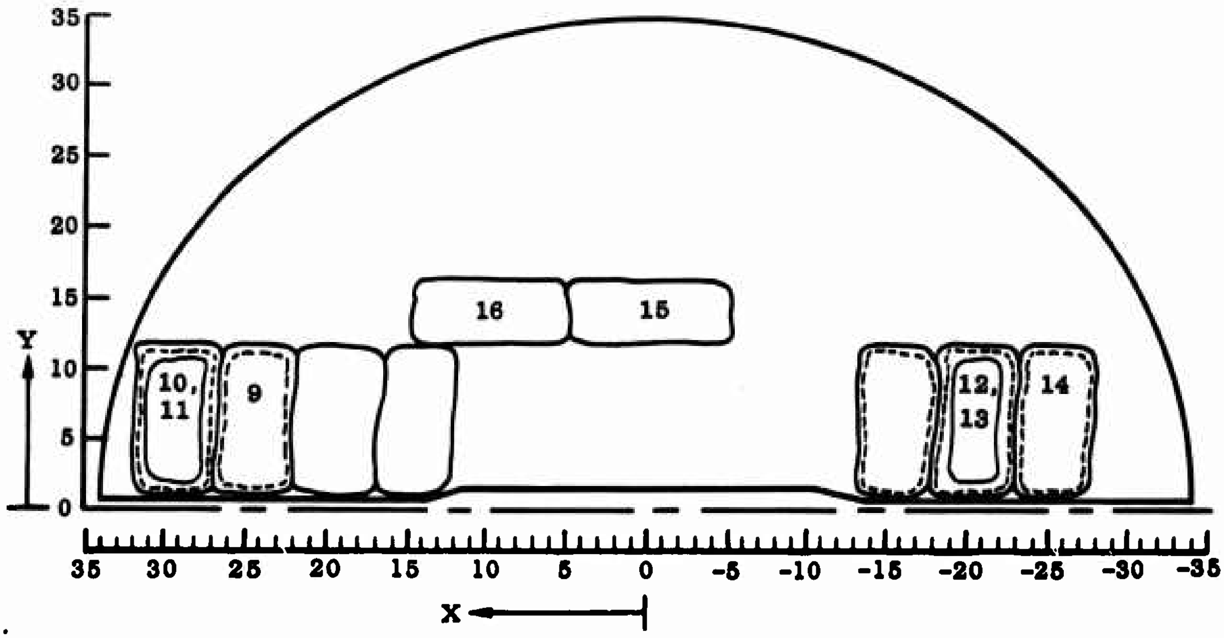
Record the following deflections for all increments of limit load. Measure the deflection relative to the jig structure.

(1) Side deflections (airplane axis), outer edge of door, 5 places; see Figure 26.
(2) Side deflections (airplane axis), fore and aft outrigger arms at hinge, and
(3) Side and vertical deflections (airplane axis), actuator arms at hinge centerline.



	BAG NO.	WT	X	WX	Y	WY	
Previous Loading							
	1	25	15	375	6	150	
	2	25	20	500	6	150	
	3	25	25	625	6	150	
	4	25	30	750	6	150	$\overline{X}$ = 1.875"
	5	25	-15	-375	6	150	$\overline{Y}$ = 6.00"
	6	25	-15	-375	6	150	
	7	25	-20	-500	6	150	
	8	25	-25	-625	6	150	
Loading No. 1 Total		200		375		1200	

Figure 22 Test No. 20 (a) - Loading, 25% Limit

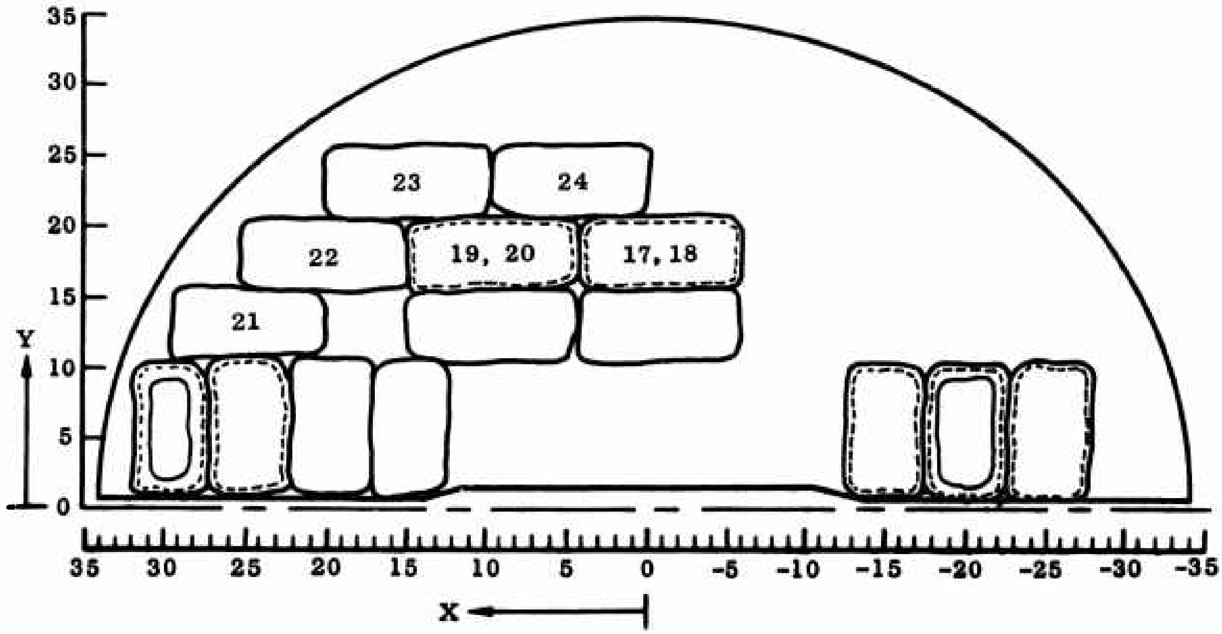


	BAG NO.	WT	X	WX	Y	WY
Previous Loading		200		+375		1200
	9	25	+25	+625	6	150
	10	25	+30	+750	6	150
	11	25	+30	+750	6	150
	12	25	−20	-500	6	150
	13	25	-20	-500	6	150
	14	25	-25	-625	6	150
	15	25	0	0	13.5	337.5
	16	25	+10	+250	13.5	337.5
Loading No. 2 Total	200	200		750		1575
Accumulative Total		400		1125		2775

$\overline{X}$ = –.8125"

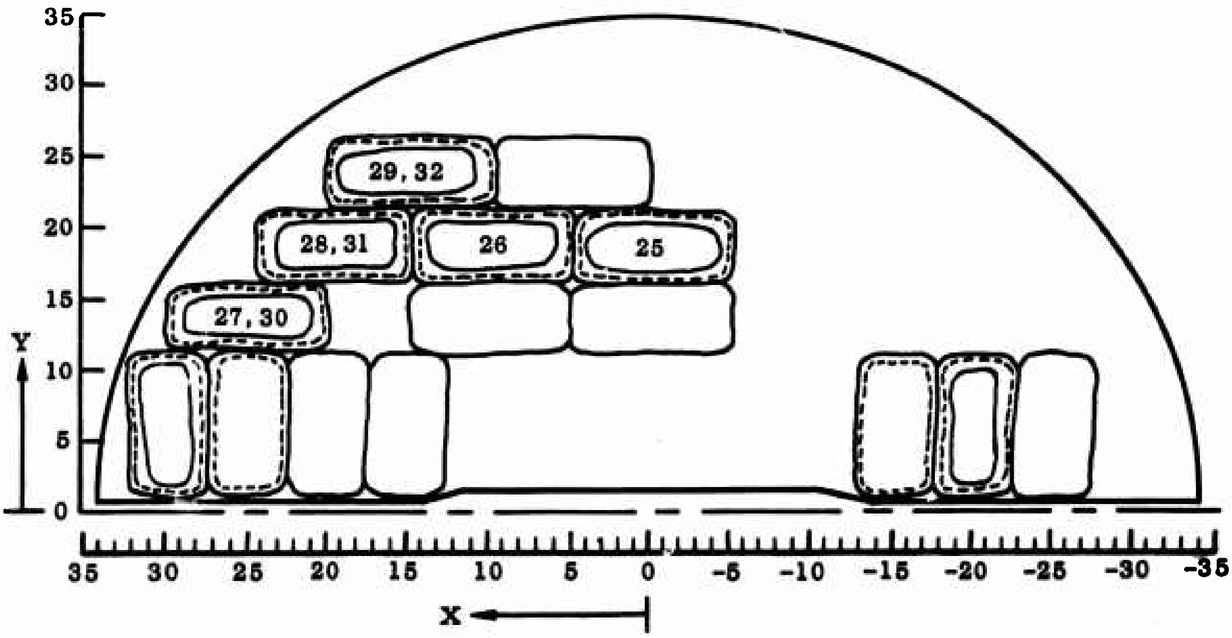
$\overline{Y}$ = 6.9375"

Figure 23 Test No. 20 (a) - Loading, 50% Limit



	BAG NO.	WT	X	WX	Y	WY	
Previous Loading		400		1125		2775	
	17	25	0	0	18.5	462.5	
	18	25	0	0	18.5	462.5	
	19	25	+10	+250	18.5	462.5	$\overline{X}$ = 5.4167"
	20	25	+10	+250	18.5	462.5	$\overline{Y}$ = 11.00"
	21	25	+25	+625	13.5	337.5	
	22	25	+20	+500	18.5	462.5	
	23	25	+15	+375	23.5	587.5	
	24	25	+5	+125	23.5	587.5	
Loading No. 3 Total		200		2125		3825.0	
Accumulative Total		600		3250		6600	

Figure 24 Test No. 20 (a) - Loading, 75% Limit



	BAG NO.	WT	X	WX	Y	WY
Previous Loading		600		3250		6600
	25	25	0	0	18.5	462.5
	26	25	10	250	18.5	462.5
	27	25	25	625	13.5	337.5
	28	25	20	500	18.5	462.5
	29	25	15	375	23.5	587.5
	30	25	25	625	13.5	337.5
	31	25	20	500	18.5	462.5
	32	25	15	375	23.5	587.5
Loading No. 4 Total		200		3250		3700
Accumulative Total		800		6500		10300

$\overline{X}$ = 8.125"

$\overline{Y}$ = 12.875"

Figure 25 Test No. 20 (a) - Loading 100% Limit

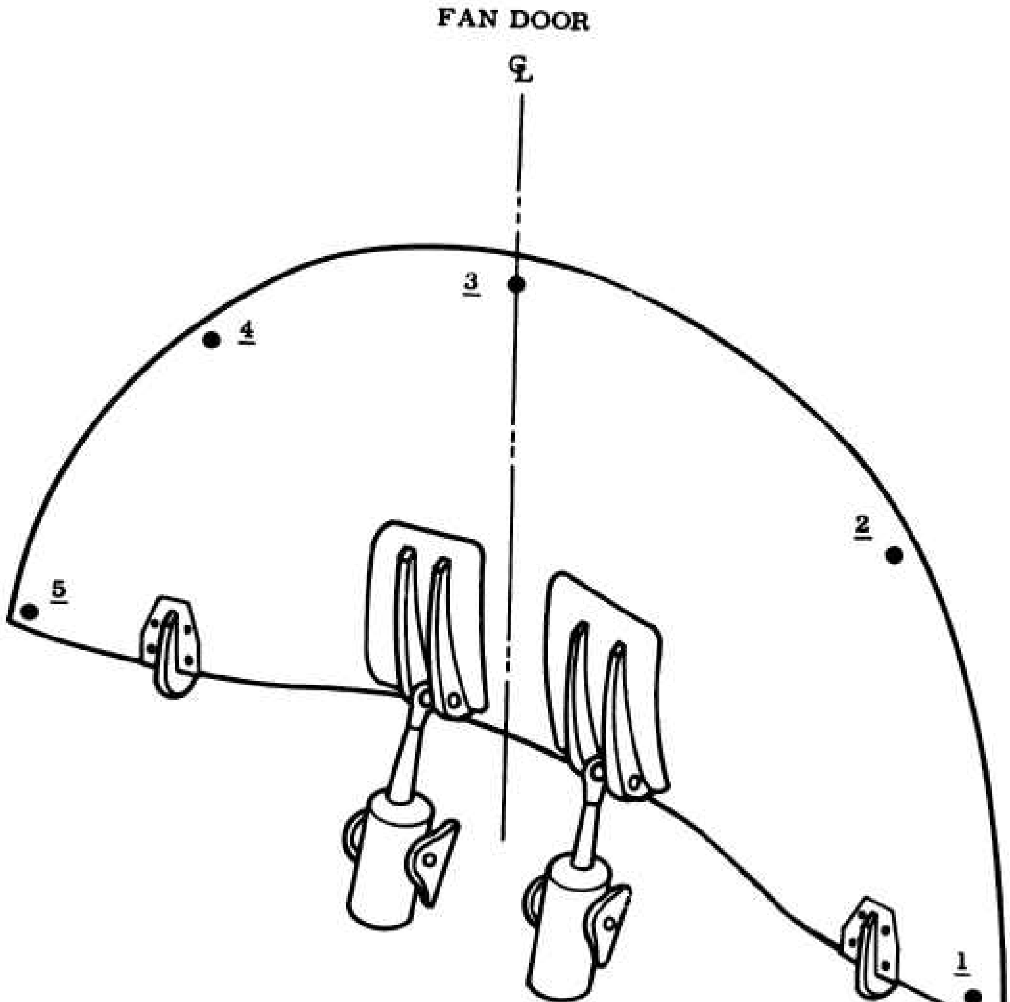


Figure 26 Approximate Points for Deflection Readings of Door, Test No. 20 (a)

TEST NO. 20 (b)

Condition

High speed flight condition F-1, ($N_z = +4.0$ g, $\theta = 0$, $\dot{\theta} = 0.146$ rad/sec, forward C.G., Mach 0.80, q = 850 p.s.f.), which causes maximum wing fan door loading with the doors closed.

Critical Structure

Wing Fan door, wing fan door actuator arms, outrigger arms, support structure, General Electric fan hub and struts.

Test Procedure

The wing fan will be mounted horizontally in a rigid jig using actual fan fittings. The door will be closed and latched as in conventional flight. The hydraulic actuators will be attached and will function during testing as described below.

Aerodynamic loading corresponding to the flight condition described above will be simulated by tension pads cemented to the external surface of the doors. Pad locations and the distribution of loading are shown in Figures 27 and 28.

The simulated aerodynamic load shall be applied incrementally and deflection measurements recorded for the following:

(1) With all hydraulic actuators functioning normally.
(2) With the outboard forward actuator and the inboard aft actuators functioning normally, and the remaining two actuators inoperative.

(3) With the outboard aft actuator and the inboard forward actuator functioning normally, and the remaining two actuators inoperative.

Instrumentation shall be provided to measure vertical reactions (aircraft axis) at forward and aft outrigger arms, and door latches.

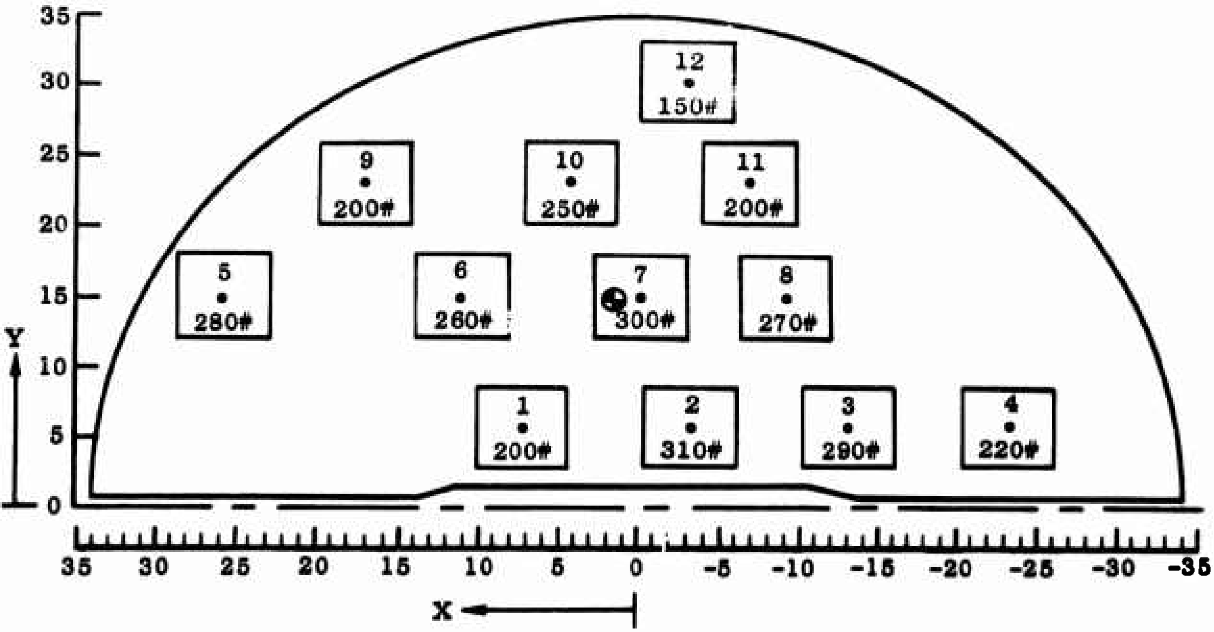
Measurements

Load measuring bolts shall be used as noted in Test No. 20 (a).

Record the following deflections for all increments of limit load relative to jig structure, unless otherwise noted:

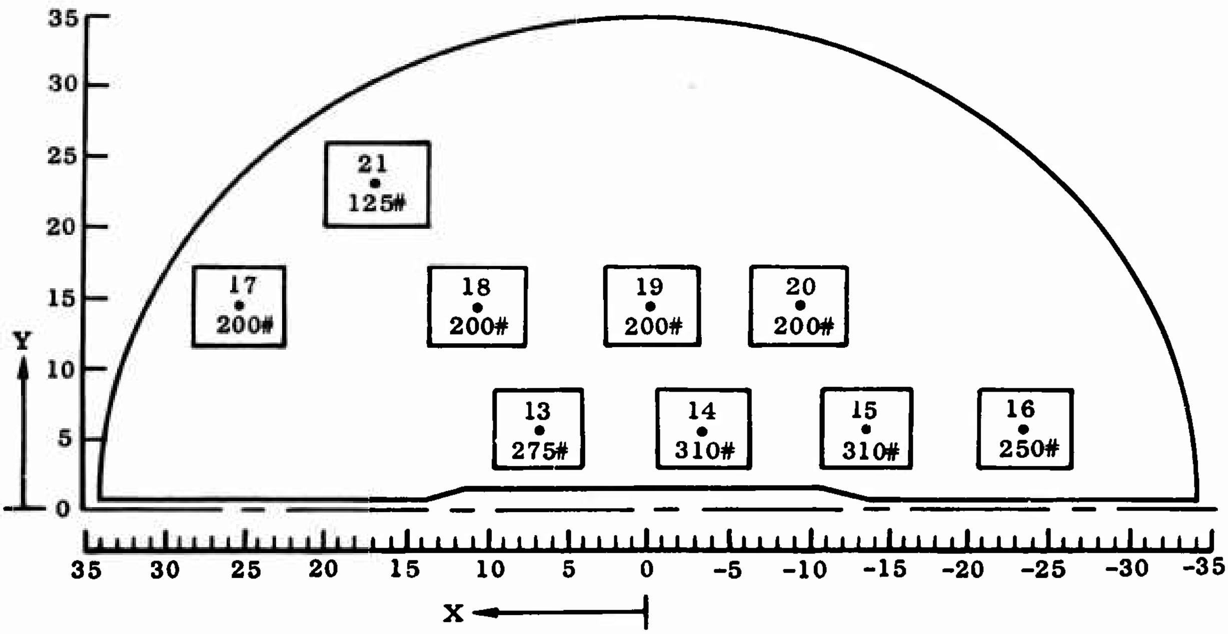
(1) Vertical deflection (A/C axis) of latches.
(2) Vertical deflection (A/C axis) of fore and aft outrigger arms at hinge centerline.
(3) Vertical deflection (A/C axis) of fore and aft actuator arms at hinge centerline.
(4) Vertical deflections (A/C axis), relative to fan bellmouth, 12 places on edge of door at the following places:

	OUTBOARD			INBOARD	
No.	Sta.	B.L.	No.	Sta.	B.L.
1	223	63	7	223	59
2	226	75	8	226	47
3	234	86	9	234	36
4	256	94	10	256	28
5	280	84	11	280	38
6	289	63	12	289	59



	PAD	T TENSION	X	TX	Y	TY	
Previous Loading							
	1	200	+ 7	+ 1400	6	1200	
	2	310	- 3	- 930	6	1860	
	3	290	- 13	- 3770	6	1740	
	4	220	- 23	- 5060	6	1320	
	5	280	+ 26	+ 7280	15	4200	$\overline{X}$ = + 0.7 IN
	6	260	+ 11	+ 2860	15	3900	$\overline{Y}$ = 14.4 IN
	7	300	0	0	15	4500	
	8	270	- 9	- 2410	15	4050	
	9	200	+ 17	+ 3400	23	4600	
	10	250	+ 4	+ 1000	23	5750	
	11	200	- 7	- 1400	23	4600	
	12	150	- 3	- 450	30	4500	
	Σ	2930		+ 1920		42, 220	

Figure 27 Limit Pad Loads for Test No. 20 (b) Outboard Door



	PAD NO.	T TENSION	X	TX	Y	TY	
Previous Loading							
	13	275	+ 7	+1925	6	1650	
	14	310	- 3	- 930	6	1860	
	15	310	-13	-4030	6	1860	
	16	250	-23	-5750	6	1500	$\overline{X}$ = -0.5 IN
	17	200	+26	+5200	14	2800	$\overline{Y}$ = 10.1 IN
	18	200	+11	+2200	14	2800	
	19	200	0	0	14	2800	
	20	200	- 9	-1800	14	2800	
	21	125	+17	+2125	23	2875	
		2070		-1060		20, 945	

Figure 28 Limit Pad Loads for Test No. 20 (b) Inboard Door

TEST NO. 20 (c)

Transition Flight, $N_z = 2.0$, $q = 45.9$ psf, $\alpha = 18^\circ$, Wing Fan Doors Closed.

Critical Structure

Wing fan doors, actuator arms, outrigger arms, support structure, fan hub and struts.

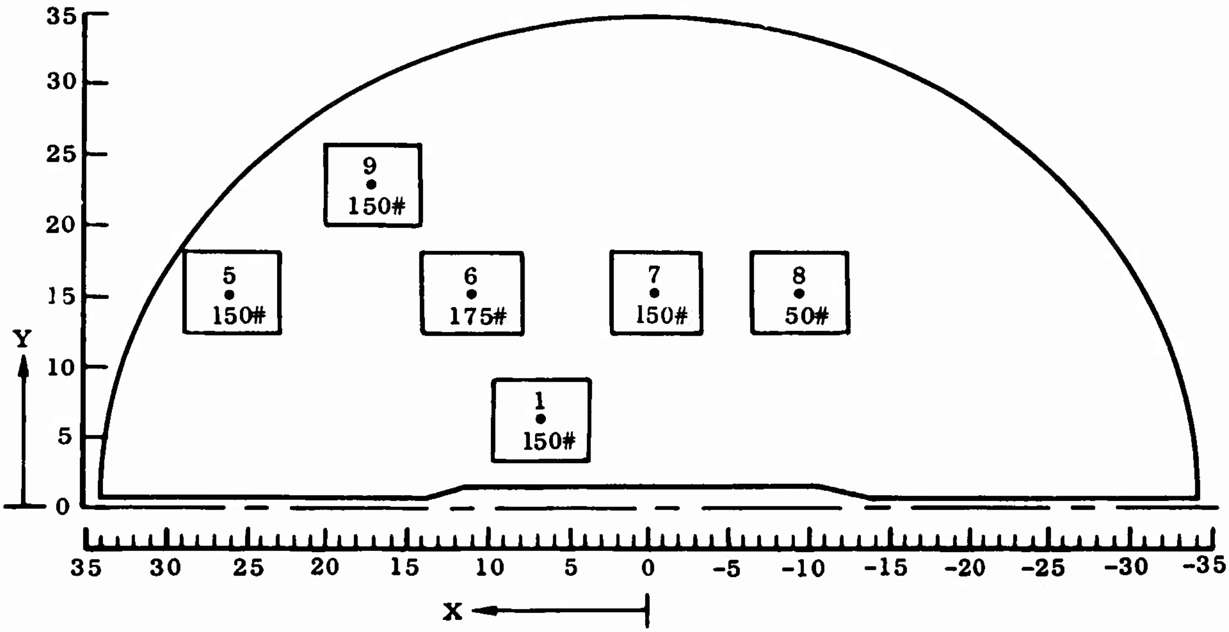
Test Procedure

The wing fan will be mounted horizontally in a rigid jig using actual fan fittings. Actual aircraft latch mechanisms and actuators will be used. The aerodynamic load shall be simulated by tension pads, which are to be located and loaded as shown in Figures 29 and 30. The load application to the pads will be such that it can be maintained while the doors open at least 4 inches at the latch location. Initially, the doors are to be closed and latched with design hydraulic pressure applied to the actuators. Increments of simulated aerodynamic load shall then be applied until limit is reached. After instrumentation readings are recorded, the doors will be unlatched and the hydraulic pressure in the actuators reduced until the doors open approximately 1 inch at the periphery. Hydraulic pressure to the actuators will then be increased to design, latching the doors. This cycle will be repeated several times with:

(1) All four hydraulic actuators operating.
(2) Only the outboard forward and inboard aft actuators operating.
(3) Only the outboard aft and inboard forward actuators operating.

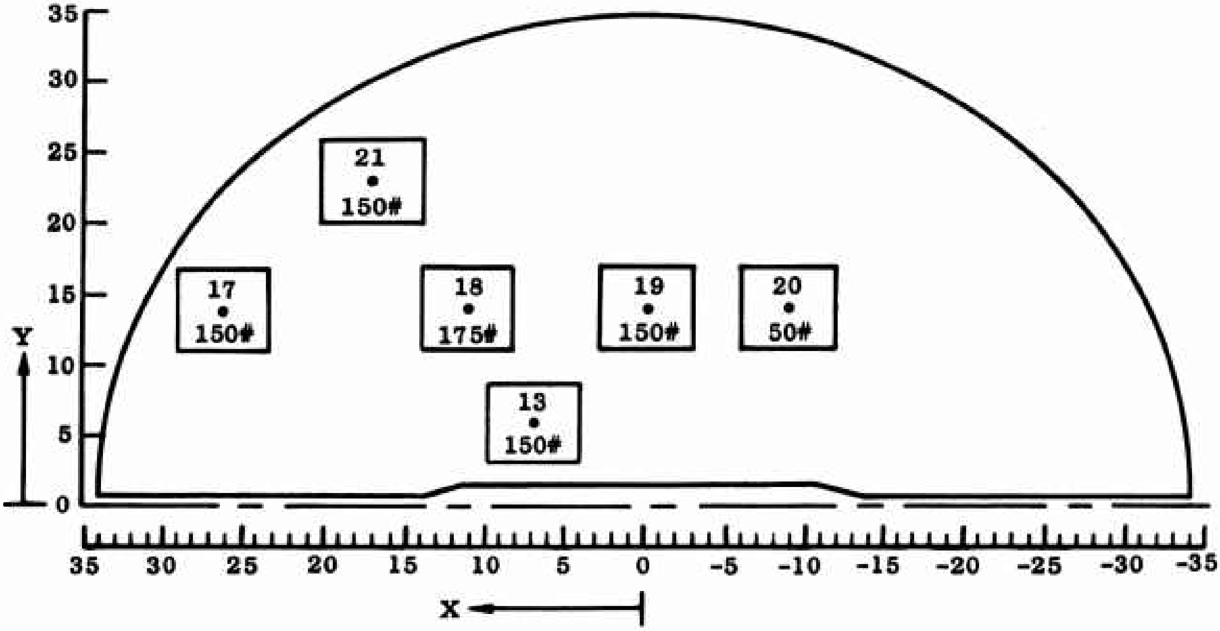
Measurements

Loading measuring bolts shall be used, as noted in Test No. 20 (a). No other deflection or load-measuring instrumentation is required for this test.



	PAD NO	T TENSION	X	TX	Y	TY	
Previous Loading							
	1	150	+ 7	+1050	6	900	
	5	150	+26	+3900	15	2250	$\overline{X}$ = 10.9 IN
	6	175	+11	+1925	15	2625	$\overline{Y}$ = 14.9 IN
	7	150	0	0	15	2250	
	8	50	- 9	-450	15	750	
	9	150	+17	+2550	23	3450	
	Σ	825		+8975		12,225	

Figure 29 Limit Pad Loads for Test No. 20 (c) Outboard Door



	PAD NO.	T TENSION	X	TX	Y	TY	
Previous Loading							
	13	150	+ 7	+1050	6	900	
	17	150	+26	+3900	14	2100	$\overline{X}$ = 10.9 IN
	18	175	+11	+1925	14	2450	$\overline{Y}$ = 14.2 IN
	19	150	0	0	14	2100	
	20	50	-9	- 450	14	700	
	21	150	+17	+2550	23	3450	
	Σ	825		+8975		11,700	

Figure 30 Limit Pad Loads for Test No. 20 (c) Inboard Door

VI. REFERENCES

Basic Ground and Full Scale Wind Tunnel Test Program.
Airplane Structural Design Criteria.
Ryan Dwg. 143F009, Sheet 6.
Ryan Dwg. 143L008.
Ryan Dwgs. 143L014 and 143L015.